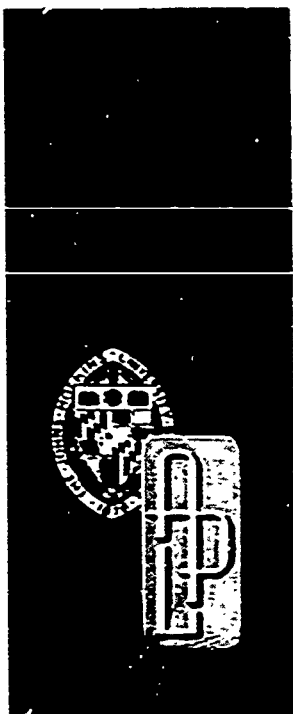


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Technical Memorandum

A SIMPLIFIED PASSIVE SPACECRAFT SEPARATION SYSTEM

by D. W. RABENHORST

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Technical Memorandum

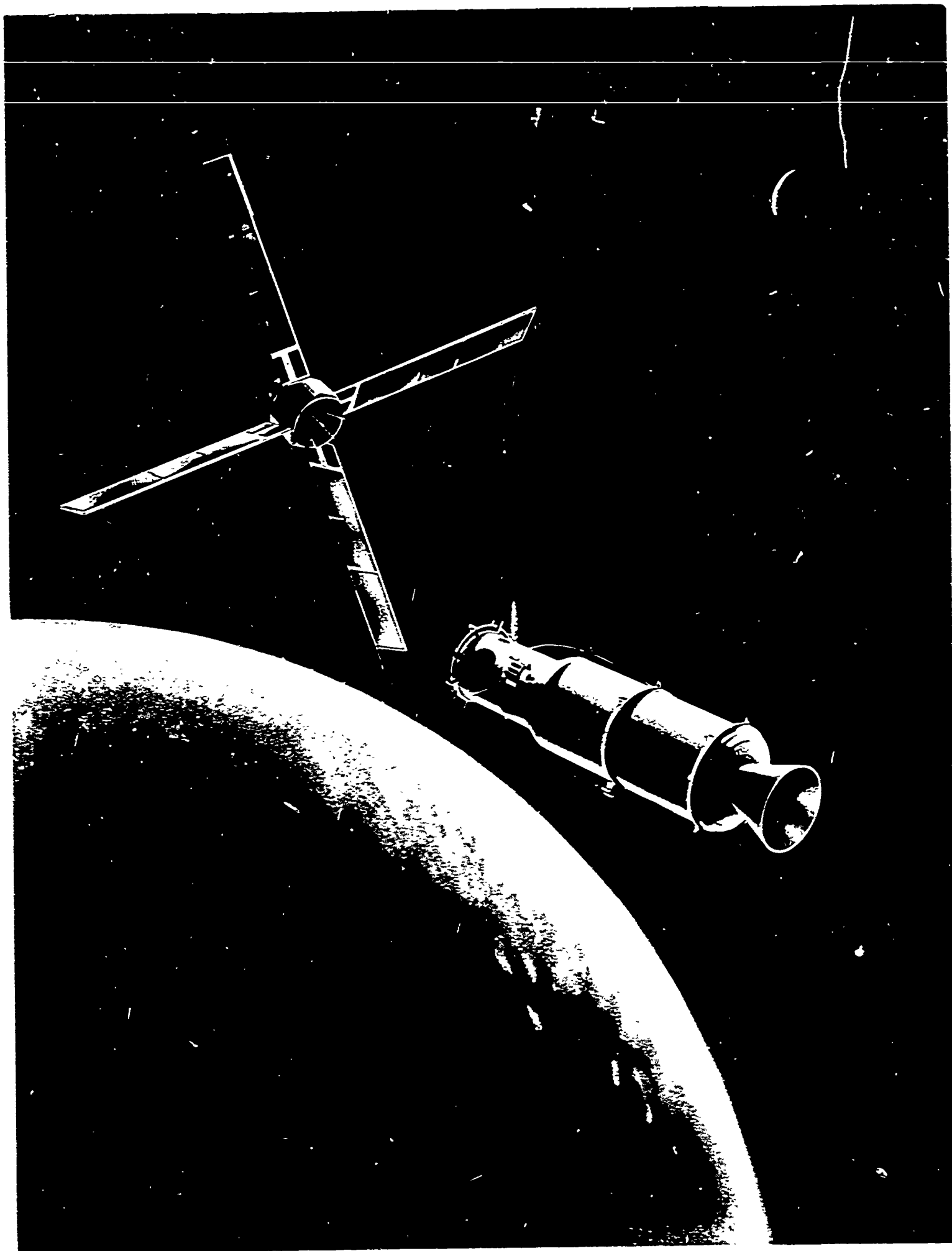
**A SIMPLIFIED PASSIVE
SPACECRAFT SEPARATION
SYSTEM**

by D. W. RABENHORST

THE JOHNS HOPKINS UNIVERSITY • APPLIED PHYSICS LABORATORY
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ABSTRACT

APL spacecraft require equipment (a) to initiate despin and unfolding of the solar blades, (b) to separate the spacecraft from the launch vehicle injection stage, and (c) to yaw the injection rocket after separation to prevent collision with the spacecraft. This report describes a unique simplified separation system which accomplishes the same objectives but has the following advantages over the method presently used: (a) the functional components are 60% lighter, (b) it has no batteries or wiring, (c) it requires no ordnance of any kind, (d) it is immune to RF static, or other electrical background disturbances, (e) its environmental temperature limitations are far in excess of similar limitations on the spacecraft and the launch vehicle, (f) it has indefinite shelf life without servicing, (g) its operation is completely independent of the launch vehicle configuration, (h) it can be operated at any place and any number of times without hazard to itself or adjacent personnel or equipment.

ACKNOWLEDGMENTS

The writer wishes to express his thanks to the following persons for their major contributions to the passive separation system.

1. John L. Letmate for the detailed design and fabrication supervision of the PDA, the trigger assembly, the PDA vacuum/shake fixture, and all of the tests on these articles; in addition to several other isolated examples of development design.
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3. Kenneth L. Nichols for design details and actual fabrication of all control rockets and rocket test fixtures, and his excellent suggestions regarding these items.
4. Laurence Whitbeck for supervision and conduct of all control rocket tests and test set-ups, test analyses, and improvisation of test equipment.
5. John P. Jones for detail design, supervision of fabrication and preliminary tests of the bolt clamp assembly, integration of the PDA, and other engineering work.
6. Robert E. Hametz for detail design and supervision of fabrication of the cable release assembly, integration of PDA, and other engineering work, including tests.
7. Peirce G. Ferriter for his enthusiastic conduct of engineering tests and major analyses, as well as for writing the Appendix to this report.
8. James T. Dunn for his detailed suggestions during the endless discussions of nearly all aspects of the new system.

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I. INTRODUCTION

The frontispiece depicts one step in the separation sequence of a typical spacecraft designed by the Applied Physics Laboratory. This general configuration has been launched fourteen times in connection with three separate spacecraft programs. The configuration is typified by the octagonal body of about 2 cu. ft. and the four long solar blades, which are literally folded around the injection stage of the launch vehicle during launching. The solar blades are held securely to blade standoffs on the rocket case by tightly wound despin weight cables. The configuration is such that by merely releasing the despin weights, the attached cables cause the spinning stage to despin and allow the spring loaded solar blades to deploy. This is the familiar yo-yo despin system used in numerous spacecrafts orbited by the United States. There are two more functions that have to be accomplished with this configuration in order to complete the post-injection-into-orbit sequence. The spacecraft must be separated from the launch vehicle, and finally, the thrust axis of the launch vehicle must be diverted so that the inevitable outgassing from the spent injection rocket, which persists for about 1600 seconds, will not cause the rocket to bump the spacecraft after separation. This last maneuver is usually accomplished (in nonspinning spacecrafts) by means of a low-impulse control rocket whose thrust axis is aligned perpendicular to the main thrust axis of the injection rocket and forward of its center of gravity.

The Applied Physics Laboratory has long been a proponent of independent separation systems for its spacecraft. In all of the cases cited, the total weight of the spacecraft separation system, including the spacecraft-to-launch vehicle structural adapter, has been consistently less than half the weight of the standard universal separation hardware otherwise provided by the launch vehicle contractor. This is largely because the structural equipment can be designed for a specific configuration instead of a variety of configurations and weights. Also, increased inherent reliability is claimed for the Applied Physics Laboratory system, since there is no functional electrical interference with either the launch vehicle or the spacecraft and since redundancy is provided in all subsystems.

The flight performance of the current APL separation system is not contested. There is no evidence to indicate that a malfunction of the current system has even occurred in flight. It is because of this excellent flight record that many persons have asked: Why do we need a new separation system? Initially, the reason was to eliminate as many of the ground handling problems as possible without sacrificing reliability. These problems have been mainly centered on the preparation of the separation system for flight and maintaining flight readiness at the launch site. On some occasions separation batteries have had to be replaced in the field after being verified for flight. And, similarly, there have been times when the sublimation timers have required repair and replacement at rather awkward times. For example, it is an extremely difficult and hazardous job to replace the sublimation switches after the spacecraft adapter has been installed on the launch vehicle for spin balancing the injection stage. This is because of the virtually inaccessible position of the sublimation switches, as is explained below.

It was reasoned that, if the sublimation timers could be replaced by an equally reliable passive timer one that would not impose severe ground handling and environmental restrictions and one that could be tested at will without destruction, the field operations with the separation system would be improved. It was further reasoned that additional simplification and improved inherent reliability could be achieved if the new passive timer could also be configured to provide initiation of the various separation functions without the requirement for batteries. Actually this requirement did not appear to be too difficult to satisfy by the use of percussion-initiated ordnance squibs, such as those in wide use throughout the world in aircraft crew ejection systems. The passive timer would be used to actuate simple gun type triggers for the pyrotechnic bolt cutters, the cable cutter, and the control rocket. However, in the configuration analysis which followed, it became apparent that it would actually be practical to design a passive separation system that, in addition to the above virtues, would not use any pyrotechnic devices at all.

This concept evolved into the simplified passive separation system described in this report.

II. THE PRESENT APPLIED PHYSICS LABORATORY SPACECRAFT SEPARATION SYSTEM

A brief description of the present separation system will be of interest before going into the details of the new simplified system. The present system is based upon the use of a sublimation timer to close the circuits of the various pyrotechnic devices in the proper sequence. The components in this system are shown in Figs. 1, 2, 3, and 4; the overall schematic diagram is shown in Fig. 5.

The sublimation timers are attached to special aluminum pads that have been previously bonded to the injection rocket head cap long before the spacecraft arrives at the launch site. The configuration and locations of these pads on the rocket are determined from prior static firing tests of similar rockets to establish the temperature-versus-time characteristics of the rocket head cap outer surface. In flight, the heat from the injection rocket is transferred to the base of the sublimation timer, and causes greatly accelerated sublimation of a solid material in the timer (usually biphenyl) to the ambient vacuum. This action allows the spring-loaded electrical contact wiper in the timer to make and break the various separation sequence circuits as follows:

First, the sublimation timer in its launch configuration provides electrical shorts across all ordnance bridgewires. At about 11 minutes after injection rocket ignition, the sublimation timer has advanced to the position where the shorts have been removed,

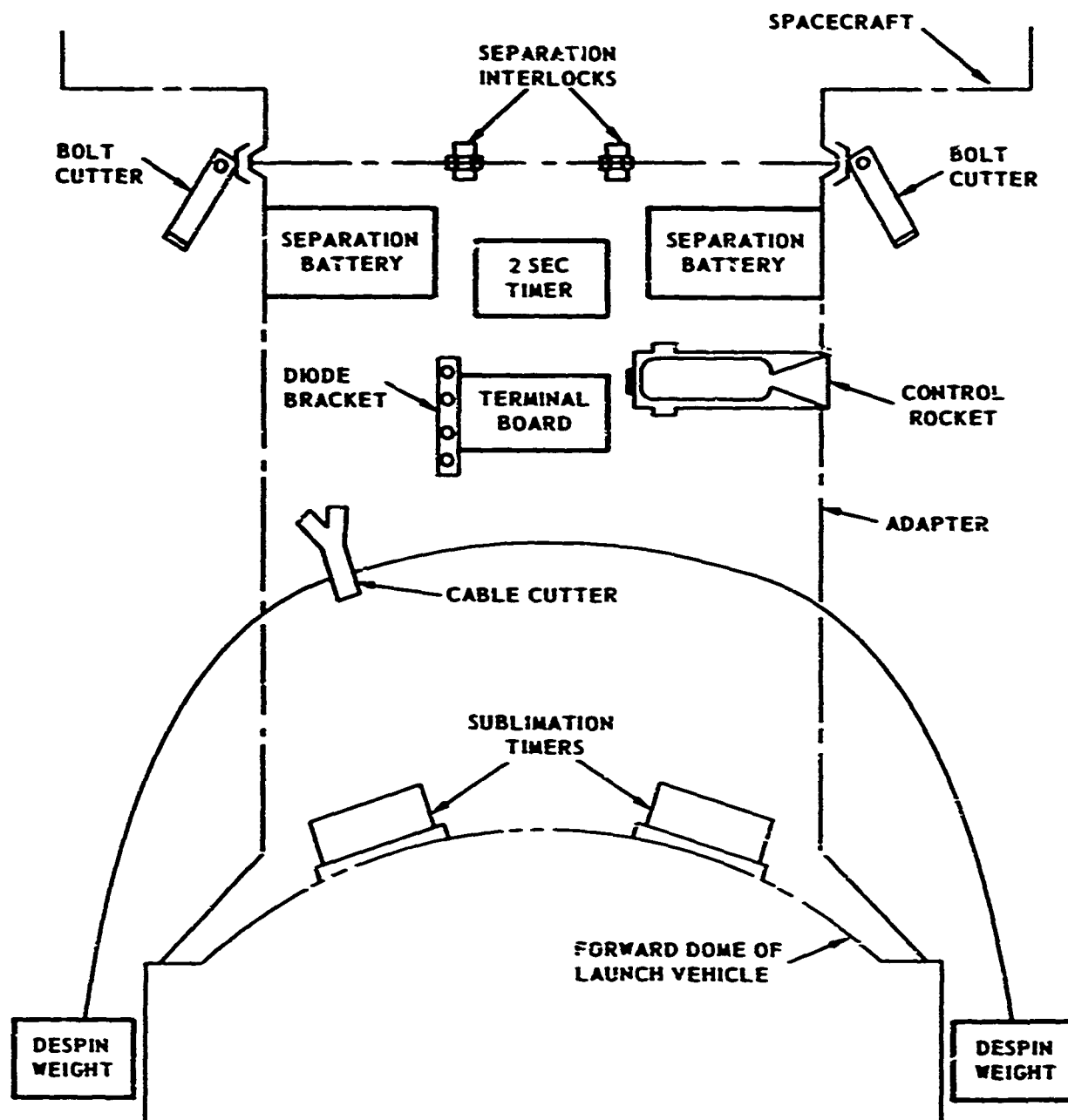


Fig. 1 CURRENT SEPARATION SYSTEM (GENERAL ARRANGEMENT)

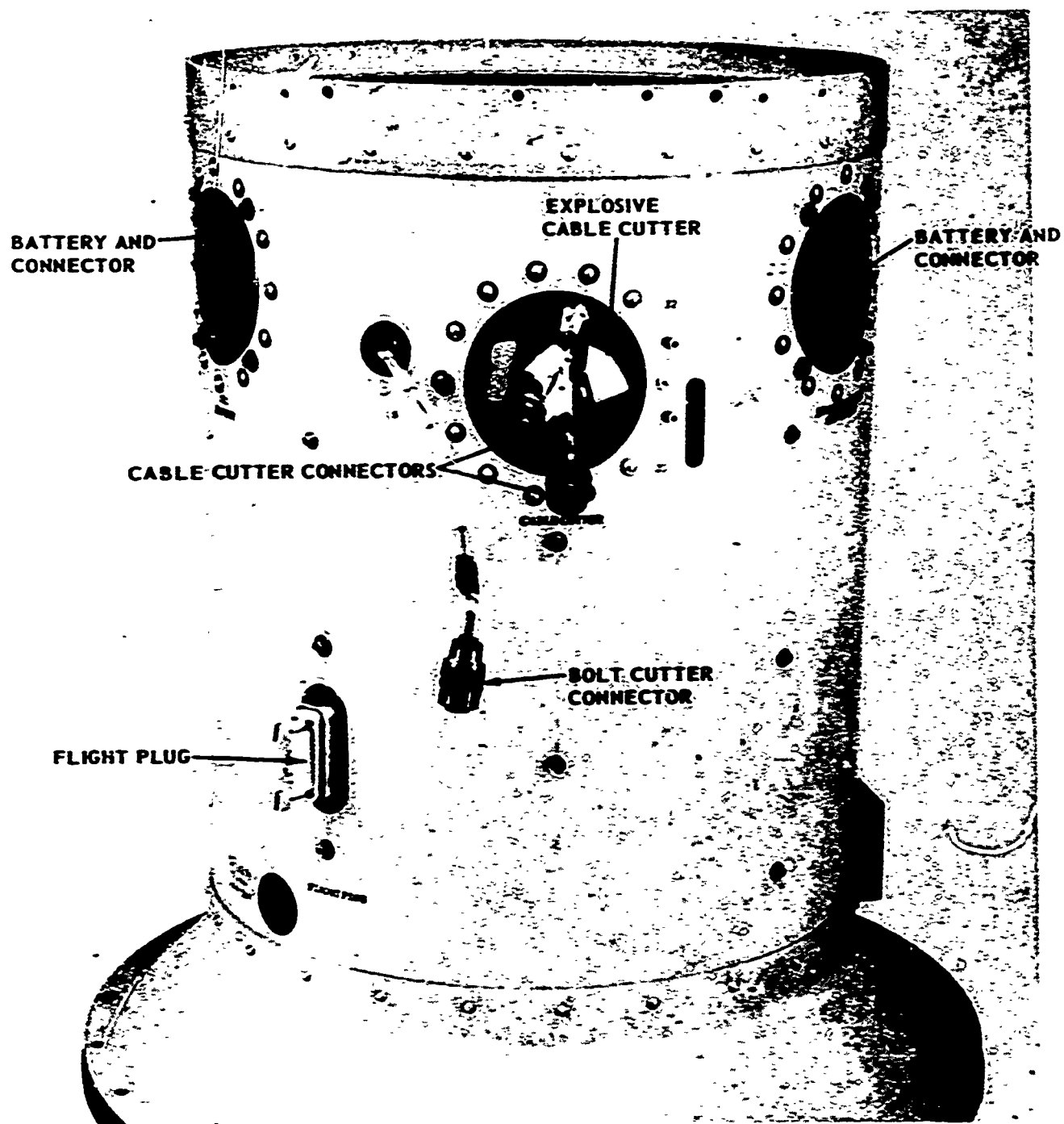


Fig. 2 CURRENT SEPARATION SYSTEM (PARTIAL VIEW OUTSIDE ADAPTER)

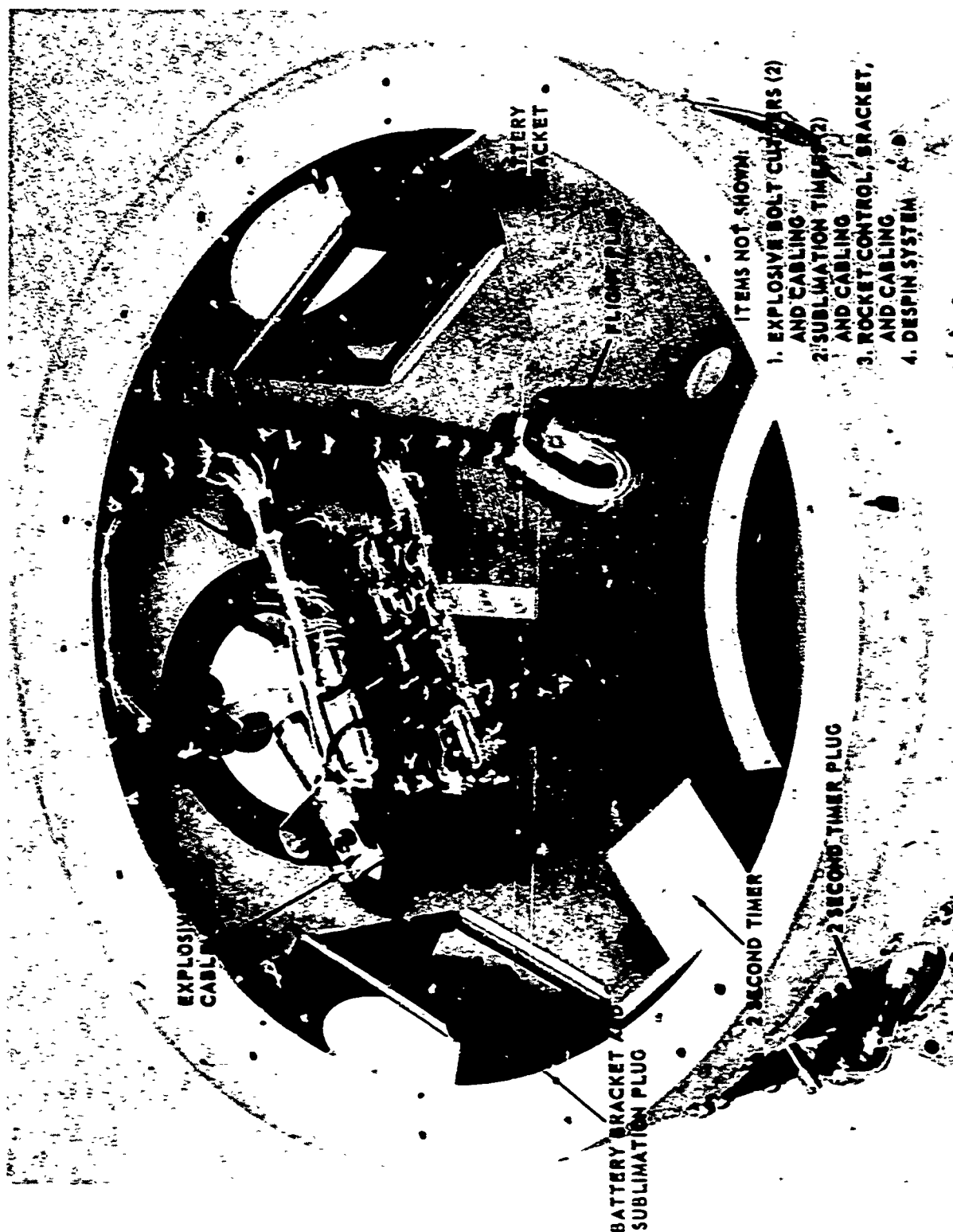


Fig. 3 CURRENT SEPARATION SYSTEM (PARTIAL VIEW INSIDE ADAPTER)

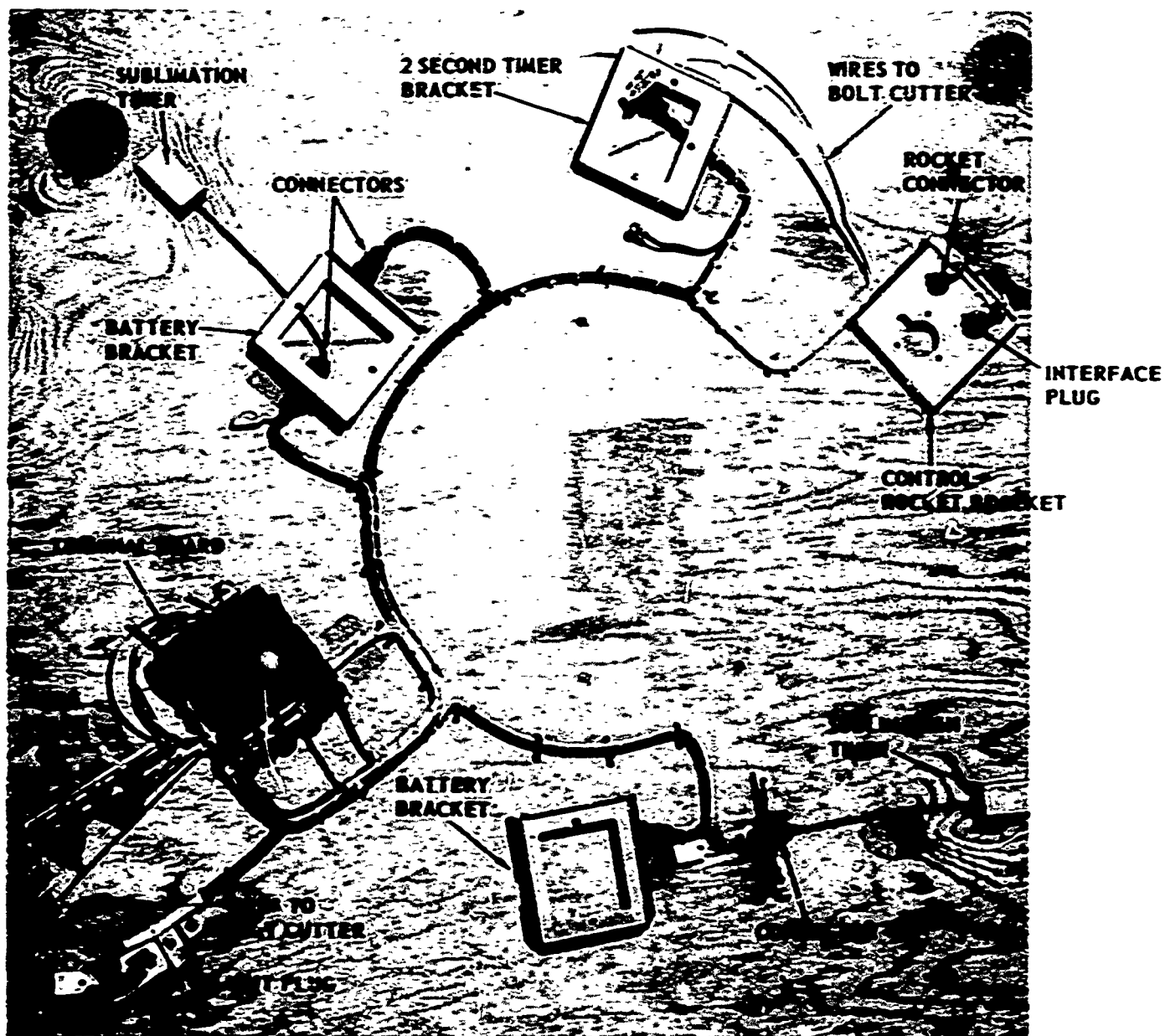


Fig. 4 CURRENT SEPARATION SYSTEM (COMPONENTS AND HARNESS)

**NOTES: NOMINAL BAT. VOLTS (0.1V)
DESPIN ACTIVATE (0.2V)
SEPARATION ACT. (0.3V)
DO NOT USE THIS SCHEMATIC
FOR WIRING PURPOSES.**

Fig. 5 CURRENT SEPARATION SYSTEM (SCHEMATIC DIAGRAM)

and current is impressed across the redundant explosive cable cutters on the despin weight release cable. The firing of the cable cutters allows nearly simultaneous release and deployment of the despin weights. The solar blades folded in trailing position against the injection rocket case are allowed to erect as the despin cables unwind. The combination of the despin weight deployment and the greatly changing polar moment of inertia as the solar blades erect causes the spin rate of the spacecraft, adapter and injection stage to rapidly decrease from a nominal 180 RPM to less than one RPM. At about 15 minutes after injection stage ignition the sublimation timer advances to the next set of electrical contacts, which closes dual explosive bolt-cutter squib circuits, causing the release of the spacecraft-to-adapter Marman clamp and allowing a push-off spring to separate the spacecraft from its adapter at a relative velocity of about 3 ft/sec. The separation causes activation of a solid-state 2 second timer in series with the small transverse control rocket. When this timer times out, the spacecraft has moved sufficiently away from the adapter so that the control rocket blast will not adversely affect the spacecraft, either from contamination or from blast forces. The firing of the control rocket causes yawing of the injection stage, so that its continuing low level thrust from outgassing of charred insulation, etc., will not allow the rocket to collide with the spacecraft after separation.

The present separation system, although reliable in flight, is sufficiently complex that it has been referred to in jest as the "second spacecraft" on the launch vehicle. Its batteries are maintained on more

or less constant charge status at the launch site right up to launch day. And its ordnance items, because of their obvious hazard, are not connected into their circuits until the last possible access time, after the spacecraft has been installed for flight and before the vehicle heat shield is installed. In addition to this, once the sublimation timers have been installed in the adapter, the temperature of the assembly must be carefully controlled so as to prevent premature operation of the timers, which would necessitate their replacement. Because of this situation, it has been necessary on some occasions to delay installation of the flight sublimation material until the launch vehicle has been installed in the air-conditioned shelter on the launch pad. It can be seen from Fig. 1 that the location of the sublimation timers on the injection rocket headcap inside the spacecraft adapter makes replacement an extremely difficult, as well as hazardous, operation.

III. THE SIMPLIFIED PASSIVE SEPARATION SYSTEM

A. General Description

The simplified separation system is truly passive. That is to say, once assembled and tested in the laboratory, it is thereafter ready for flight; its readiness can be confirmed at any accessible time by visual inspection. And, once launched, it has the capability of providing automatically the necessary separation functions in the proper sequences with no functional interface with either the spacecraft or the launch vehicle. The following advantages are also claimed:

1. Its components are 60% lighter than their functional equivalents in the present system.
2. It has no batteries, wiring, or electrical components
3. It has no pyrotechnic devices of any kind. Hazard is negligible.
4. It is immune to RF, static, or other electrical background disturbances.
5. It has no ground environmental temperature limitations. It can survive any temperatures normally expected to be encountered in ground handling and transportation such as the typical ground environment design limits of from -60°F to +160°F.
6. It has indefinite shelf life even in the launch-ready configuration.

7. Its operation and design are completely independent of the launch vehicle configuration. It has no functional or physical interface with the vehicle.
8. If desired, it can be completely tested at any convenient number of times or places (including on the launch vehicle) without hazard to itself or adjacent personnel or equipment.
9. The system design is flexible. Any of the subsystems, the passive delay actuator, the trigger assembly, the cable release assembly, and the control rocket, can be integrated separately or collectively into existing spacecraft separation systems or other mechanical systems as desired.
10. The system is accurate. Timing sequences are typically repeatable within $\pm 2\%$ as compared to $\pm 20\%$ for the present sublimation timer system. Although high accuracy is not a stringent requirement in the present application, it is clear that possible future applications involving longer times (many hours) and more mechanical functions would require much greater accuracy than is currently available.

Figure 6 illustrates the various components that make up the simplified passive separation system. The simplicity of the system is illustrated, in part, by the fact that, although Fig. 6 is labeled a "block diagram," it is also an accurate schematic of the system. The principal subsystem is the passive delay actuator/trigger assembly, which is actually a pneumatic timer capable of performing work at discrete times after being launched into the vacuum of outer space. The sole input to the PDA is the exposure to the vacuum almost immediately

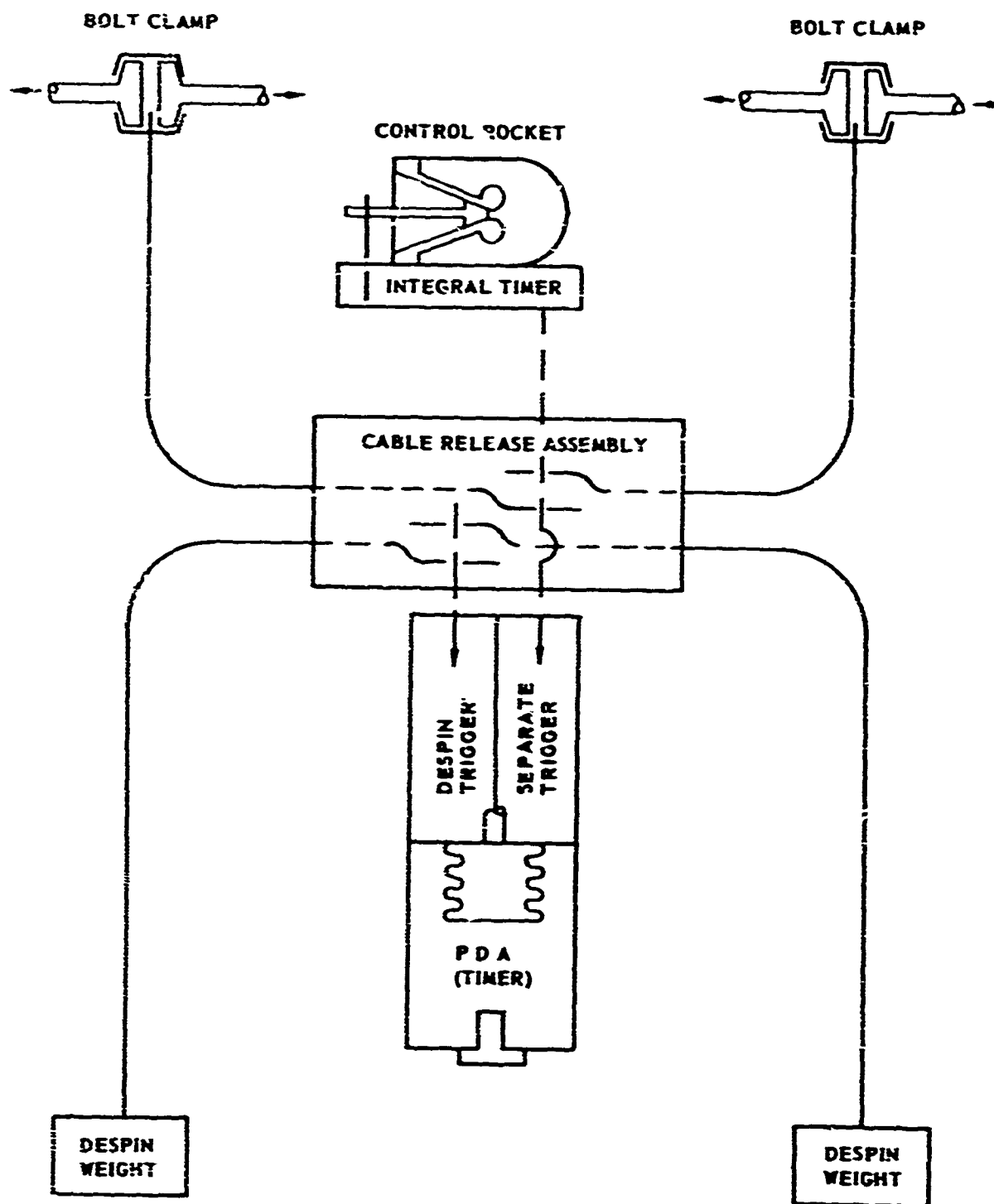


Fig. 6 PASSIVE SEPARATION SYSTEM BLOCK DIAGRAM

after the launch vehicle lifts off the launch pad, while the output of the integral trigger unit is the 23 pound snap action of each of the two triggers at specified times after lift-off. The first trigger causes the simultaneous release of the two despin weight release cables, and the second trigger causes release of two similar cables which secure special bolt clamps on the main spacecraft separation clamp.

It is interesting to note that with this arrangement both the despin weight assemblies and the spacecraft separation clamp can accommodate the new system without redesign. The despin weights are used as is, and their release cable is modified to permit release by unlatching, instead of by explosive cable cutter as in the present system. Only the retaining bolts on the separation Marman clamp are changed; the new bolts are designed to be unlatched instead of being cut by explosive bolt cutters as in the present system.

Finally, the action of the second trigger also fires the control rocket through an integral 2 second mechanical delay timer. The timer prevents the control rocket from firing before the spacecraft has adequately cleared the adapter during the separation process. The control rocket is a unique development in that it contains no pyrotechnic elements. The impulse is derived from the explosive evaporation of an appropriate fluid when suddenly exposed to the vacuum of outer space, as will be explained below.

Referring to Figs. 7 and 8, it is noted that all of the passive separation system components can be located on the outside of the spacecraft adapter for maximum accessibility and ease of visual inspection. All of the functioning components of the system are securely locked by the trigger shafts upon assembly, and remain in this condition until the triggers are armed and fired during the launch cycle. The trigger

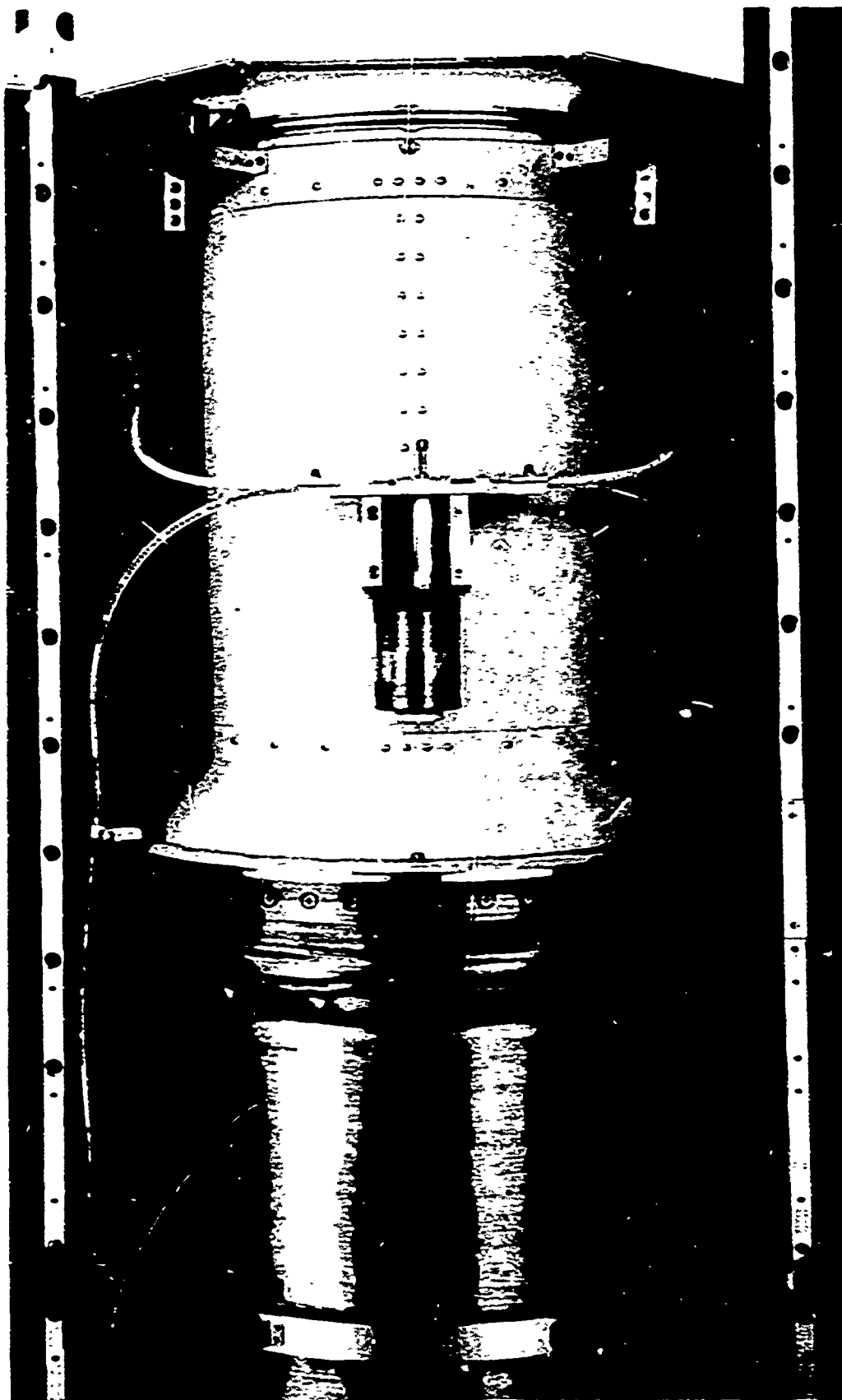


Fig. 7 PASSIVE SEPARATION SYSTEM FLIGHT INSTALLATION (FRONT VIEW)

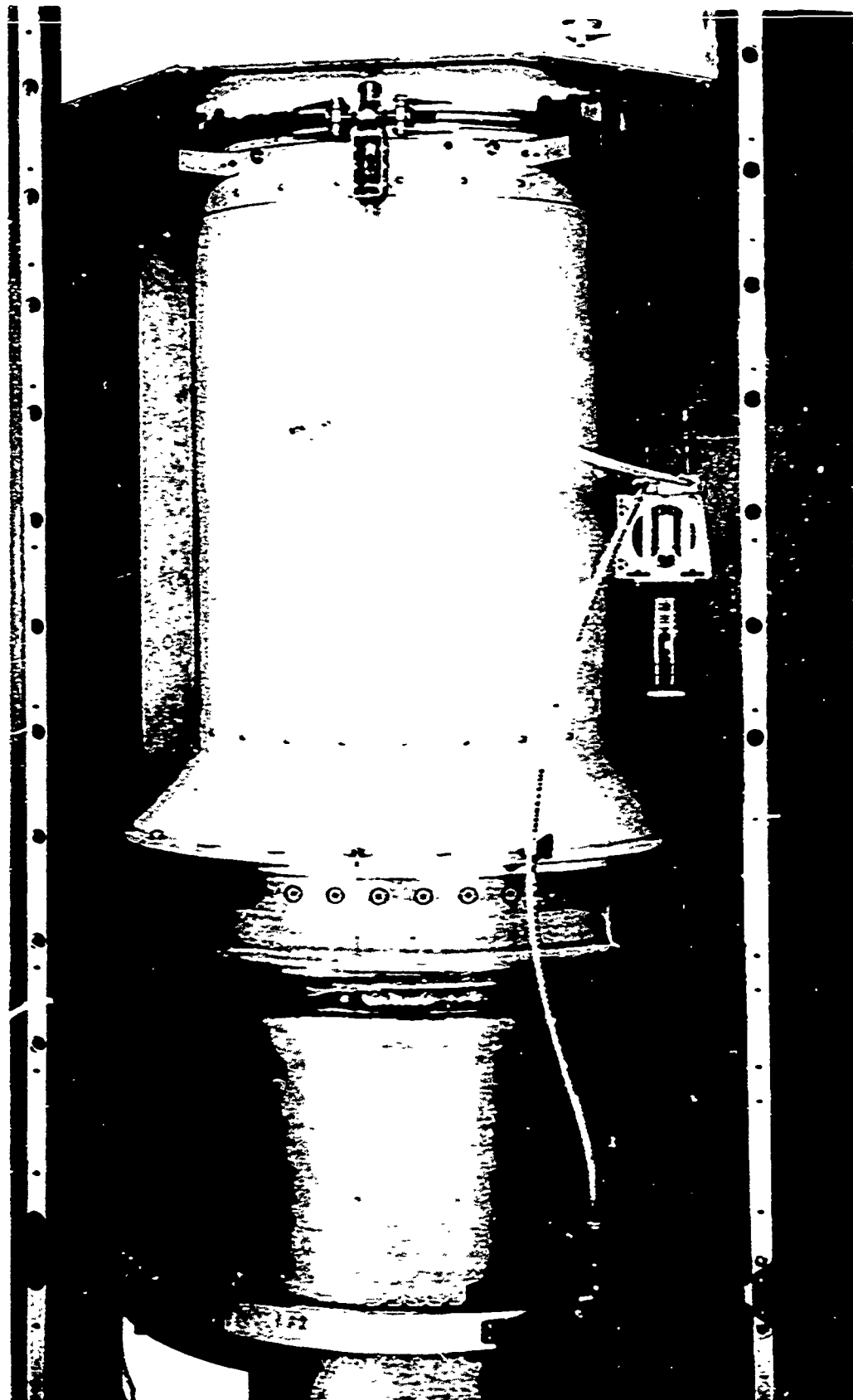


Fig. 8 PASSIVE SEPARATION SYSTEM FLIGHT INSTALLATION (SIDE VIEW)

shafts, in turn, are not physically connected to the passive delay actuator until the latter unit is automatically armed shortly after launching. If it is desired to accomplish final assembly at the laboratory, prior to shipping to the launch site, then it will be required that a nonflight safety pin be installed in the PDA to prevent inadvertent operation during air shipment. On the other hand, if it is desired to accomplish final assembly at the launch site and subsequently demonstrate flight readiness, this can be readily accomplished any desired number of times by means of the small attaché-case test kit described below.

The weights of the various system components are illustrated in Fig. 9, where a comparison is made with the equivalent components in the present system.

B. Detailed Description

1. The Passive Delay Actuator (PDA)--Figure 10 is an early photo of the PDA, and Fig. 11 illustrates the components used in the assembly. Operation of the PDA is quite simple, as indicated in the functional diagram, Fig. 12. Volume #1 is a metal bellows welded to the output shaft plate at one end and welded to the outer case end plate at the other end, so as to provide an absolute seal between the bellows and the outer case, which defines Volume #2 in the figure. The inside of the bellows is vented to ambient however. Volume #3 is included as a safety device, and is not a mandatory feature of the PDA. Its purpose will be explained later. And, similarly the spring inside the bellows is not a mandatory feature, since the thickness of the bellows material can be made sufficient to provide the required spring force. However, without going to considerable expense, the spring rate of a standard bellows cannot be controlled to much better than $\pm 20\%$ of

ITEM	PRESENT SYSTEM	PASS. SEP. SYS.
	Wt/lb	Wt/lb
1. Adapter structure, plus all required cabling, brackets and steel doublers for access holes, cable cutter and 2 sec. timer	6.43	3.55**
2. Batteries (2)	0.50	None
3. Bolt cutters or clamps (2)	0.20	0.30
4. Sublimation timers, including mounting pads (2)	0.94	None
5. Control rocket	0.12	0.10
6. Trunnion bolt (2)	0.05	(Part of 3)
7. Despin cable assembly	0.14	0.10
8. PDA/trigger assembly, including brackets	None	0.56
9. Cable release assembly	None	0.42
10. Bolt camp cable assembly	None	0.06
Total weight of separation system =	8.38	5.09

NOTES:

- * Items common to both systems, such as Marman clamp, despin weights, release mechanism and blade tie-down straps not included (5.69 lb.).
- ** Weight of adapter alone. Other items, including access holes and doublers, not required.

Fig. 9 WEIGHT COMPARISONS - PRESENT AND PROPOSED SYSTEMS*



Fig. 10 PASSIVE DELAY ACTUATOR

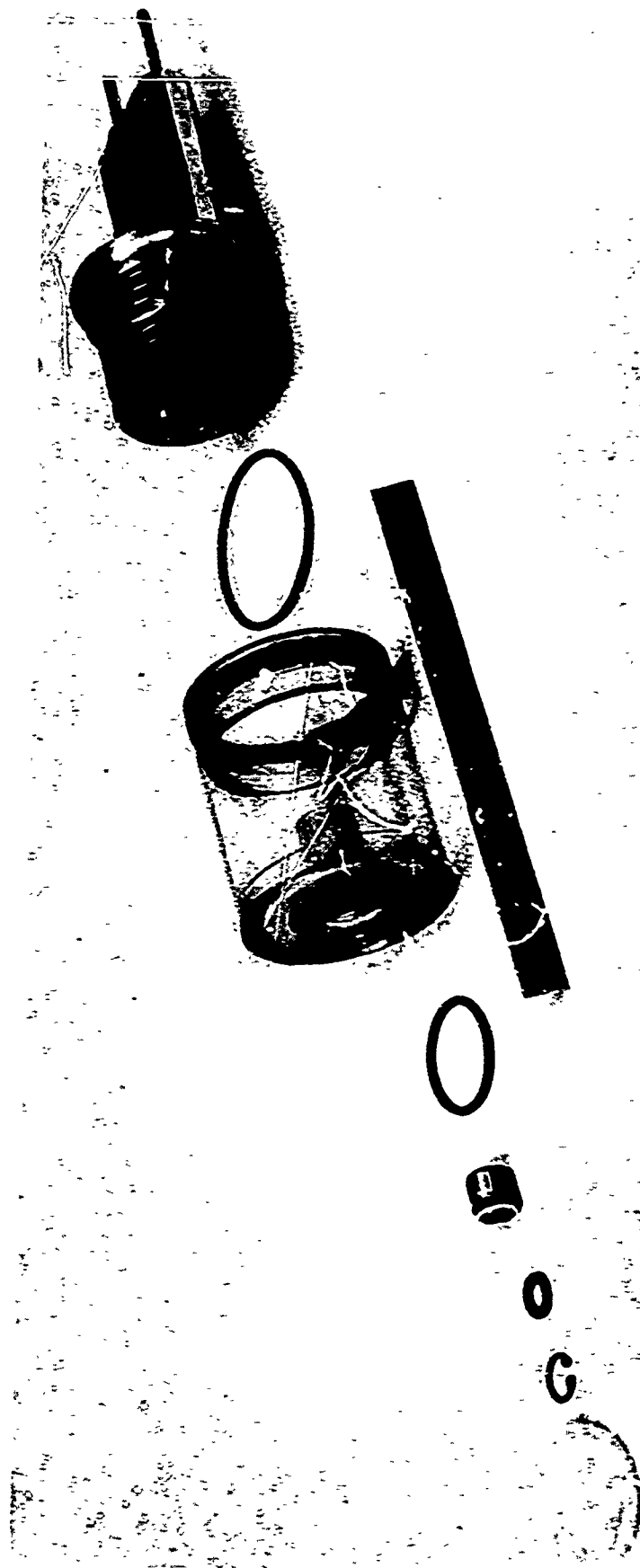
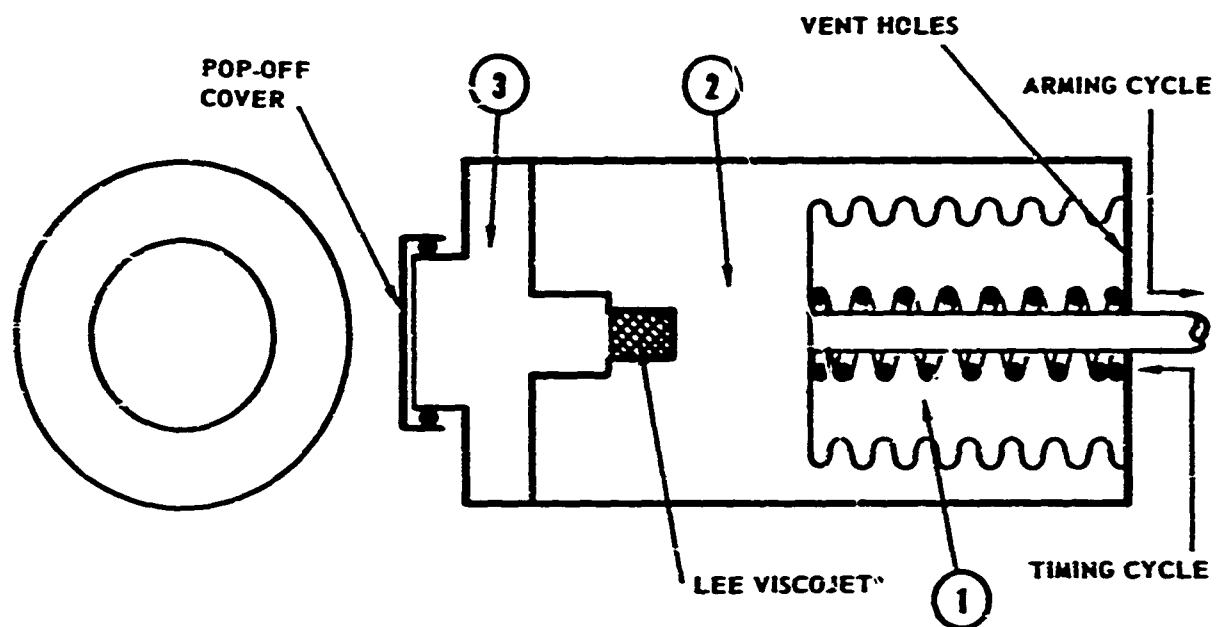


Fig. 11 PDA-EXPLODED VIEW (TRIGGER ASSEMBLY ALSO SHOWN)



*PROPRIETARY DEVICE MANUFACTURED BY THE LEE CO., WESTBROOK, CONN.

Fig. 12 PASSIVE DELAY ACTUATOR-FUNCTIONAL DIAGRAM

theoretical, whereas the spring rate of the inexpensive compression spring can be controlled to very close tolerances.

When the PDA is on the ground, all three internal volumes are at the ambient pressure, about one atmosphere. However, when launch vehicle lift-off occurs, the ambient pressure rapidly approaches zero, as does the pressure in Volume #1, which is vented to ambient. But prior to this two things happen. First, when the ambient pressure drops to about 13 psia, the differential pressure across the pop-off cap is sufficient to provide the force necessary to deploy the cap. The main purpose of the cap is to act as a dust cover during all ground handling operations with the PDA. It could very easily be removed manually at last access to the spacecraft as nonflight hardware. However, as a redundant safety feature, the cap is designed to pop-off in flight without noticeably affecting the timing performance of the PDA. Should there be a leak in the cap that would be large enough to bleed off the air in Volume #3 without developing the force necessary to deploy the cap, then the configuration (the size of Volume #3) is such that the timing accuracy of the PDA is virtually unaffected by the leak in the cap. In other words, the size of Volume #3 is selected so that if the leak is small, adequate differential pressure to deploy the cap is reached before Volume #3 bleeds down. And, if the leak is larger, then the size of the hole is greater than the effective bleed hole in the Lee Viscojet, which is the primary metering device in the PDA.

The second thing that happens after lift-off is that at about 7 psia ambient pressure (about 20,000 feet altitude) the differential pressure across the bellows shaft plate is sufficient to cause a force in excess of the combined spring forces of the bellows and spring, at which time the bellows shaft moves rapidly to the right against a stop. The PDA

is now armed, and will always seek to return to its prelaunch position no matter what happens, including a catastrophic leak anywhere. The maximum force available for arming is about 50 pounds, whereas the maximum spring return force is about 10 pounds.

Naturally, when the pop-off cap is deployed, the pressure in Volume #3 goes immediately to the ambient, and the pressure drop is essentially one atmosphere. It is not exactly one atmosphere, since the pressure drops slightly when the bellows shaft plate moves to the right. The Lee Viscojet is designed to bleed the air from Volume #2 at a precise rate under these conditions. When the pressure in Volume #2 drops below that which will produce a force balancing the total spring (and friction) forces on the bellows shaft and shaft plate, the shaft begins to move very slowly and very smoothly. The system volumes, areas, spring rates, and bleed rates have been carefully selected so as to hold the shaft on the stop until about 4 minutes after the launch vehicle has ceased thrusting and the spacecraft is in orbit. This means that the only moving part in the PDA (the shaft plate and its integral output shaft) is locked up tightly against its stop with an average of about 6 pounds force during almost the entire launch flight when the physical environment is at its extreme levels. Then, in the absolute calm of orbital environment, it times out with precision.

It is clear that this demonstrated precision requires an accurate metering system, and one that is insensitive to normal handling contamination, normal humidity changes, and normal assembly and testing procedures. Early tests, with sintered stainless steel, compressed metal screens, and porous ceramic, while demonstrating the basic feasibility of the PDA, also demonstrated the futility of the porous material approach to precise air metering. It was soon found

that a misplaced finger print, a cloudy day, or other similar situations, including just plain testing, would cause timing changes of 20% or more. This was not surprising since the bleed hole sizes were measured in microns. At this point it was suggested that a Lee Viscojet might work, and this proved to be the solution to the metering problem. The basic functioning of the Viscojet is shown in Figs. 13 and 14. In this device fluid (in this case air) motion through the Viscojet is the result of the differential pressure across the Viscojet. Kinetic energy is removed from this fluid by the repeated acceleration and deceleration over and over again as the fluid passes from one spin chamber to the next in the Viscojet. As the energy is removed, the flow is slowed down to the extent that the actual flow through the Viscojet exhaust hole is as though the hole were many, many times smaller than it is.

The PDA uses a Viscojet having approximately 400 spin chambers and a minimum passage diameter of 0.005 inch. It is protected upstream and downstream by filters having more than 100 holes of about 0.004 inch diameter, so the filter open area is hundreds of times greater than the effective hole in the Viscojet, which is about 0.0005 inch. The total size of the Viscojet used in the present PDA is about 9/16 inch diameter by less than 1/2 inch long. It meters about 6 cubic inches of air at an average differential pressure of about 5 psi in 24 minutes.

Other Viscojet configurations being designed for future applications will be capable of metering 1 cubic inch of air in 50 hours, and still have a minimum passage diameter of 0.005 inch.

Whenever the PDA is returned to one atmosphere ambient pressure, such as after a test or after a demonstration, it returns automatically to the prelaunch condition. It can be tested any desirable

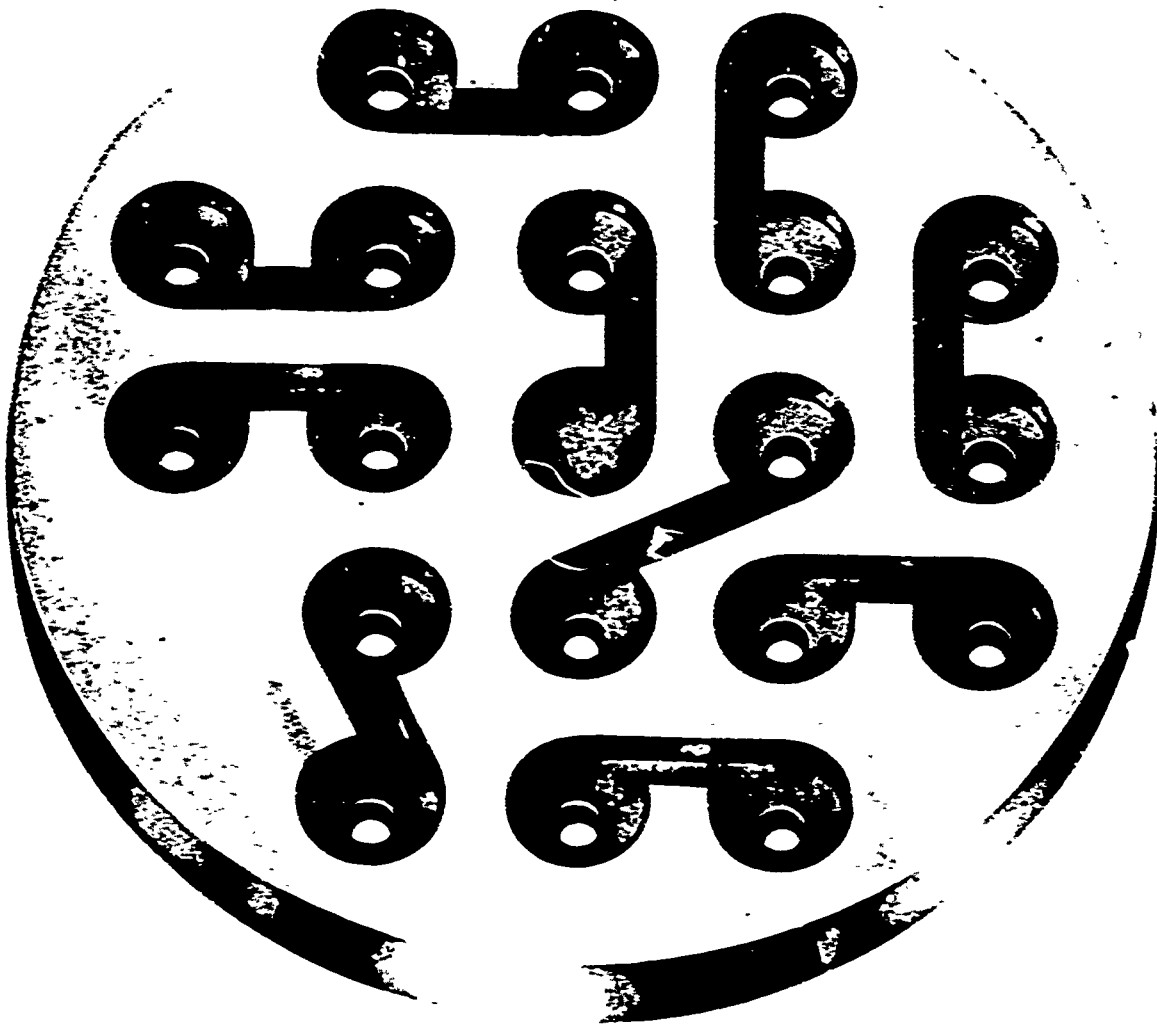
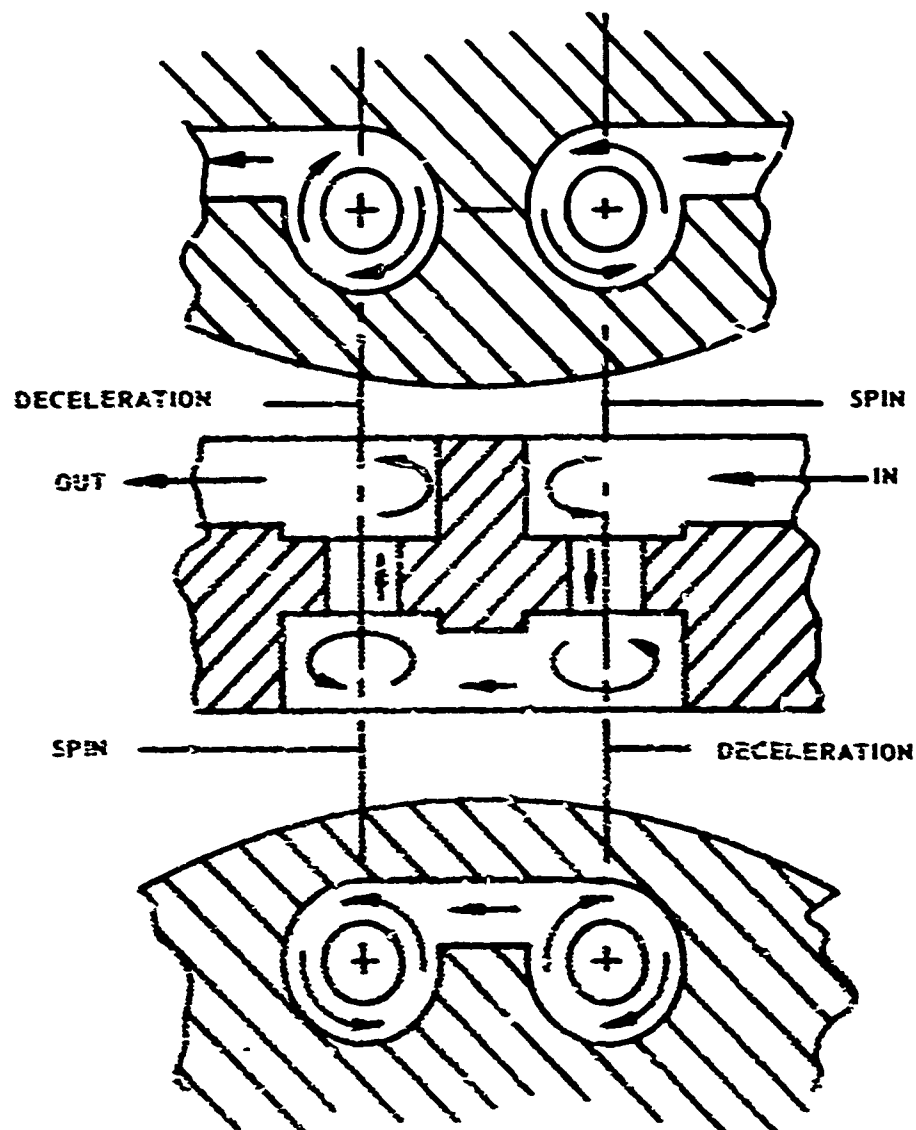


Fig. 13 LEE VISCOJET GENERAL ARRANGEMENT



- A. ABOVE ARE THREE CROSS SECTIONS OF A PORTION OF A VISCO DISC. THEY SHOW HOW THE FLOW REPEATEDLY PASSES THROUGH THE SAME DISC. THE FLOW PATH BEGINS ON ONE SIDE AT THE CENTER OF THE DISC.
- B. THE EXIT SLOT FROM EACH DECELERATION CHAMBER IS IN A DIRECTION WHICH IS OPPOSITE TO THE DIRECTION OF SPIN. THIS FORCES THE LIQUID TO COME TO REST BEFORE IT MAKES ITS EXIT FROM THE DECELERATION CHAMBER.
- C. ON THE SMALLER SIZES EACH VISCO DISC IS FABRICATED FROM THE THREE PHOTO ETCHED PLATES. ON THE LARGER SIZES THIS DISC IS OF ONE PIECE CONSTRUCTION.
- D. THE VISCO DISC IS COVERED BOTH TOP AND BOTTOM WITH A FLAT LAPPED DISC. (NOT SHOWN)
- E. THE VISCO DISCS AND THEIR COVERS ARE RIGIDLY AND PERMANENTLY CLAMPED IN THE CARTRIDGE.

Fig. 14 LEE VISCOJET PRINCIPLES OF OPERATION

number of times without degradation. The bellows are designed for 100,000 cycles without failure. Ordinarily, in the laboratory the tests are conducted merely by subjecting the PDA (and its integral trigger assembly) to a vacuum and recording the times of the (trigger) events. However, there are times when this is not practical. An example of this is illustrated in Fig. 15, where the PDA is shown undergoing a functional test in a vacuum environment while at the same time being subjected to prototype vibration levels. In this case a small vacuum pump was used in conjunction with the pressure-tight shake fixture to provide a unique and very useful test medium.

There are, of course, other situations where it would be necessary to operate the PDA, but under conditions where the vacuum environment is not practical, such as during a vibration test of the entire spacecraft, or, possibly, for functional tests of the PDA on the launch vehicle or the launch pad. Equipment designed to permit this type of testing is shown in Fig. 16. This equipment, which occupies a space no larger than an attaché case is capable of running about 1000 tests of the PDA without recharging. It uses a miniature high-pressure nitrogen bottle with appropriate pressure regulators to charge Volume #2 of the PDA to a pressure higher than the 1 atmosphere ambient pressure. When this equipment is removed, the PDA bleeds from the higher pressure down to the ambient in the same manner that it bleeds from 1 atmosphere down to zero ambient pressure in flight, but faster. This equipment has also been used many times for lecture demonstrations of the PDA timing accuracy.

2. The Trigger Assembly--It was explained above that the PDA output shaft extends during the arming cycle, and slowly retracts again during its timing cycle with a potential force of considerable magnitude. Initially it was considered that this force would be adequate

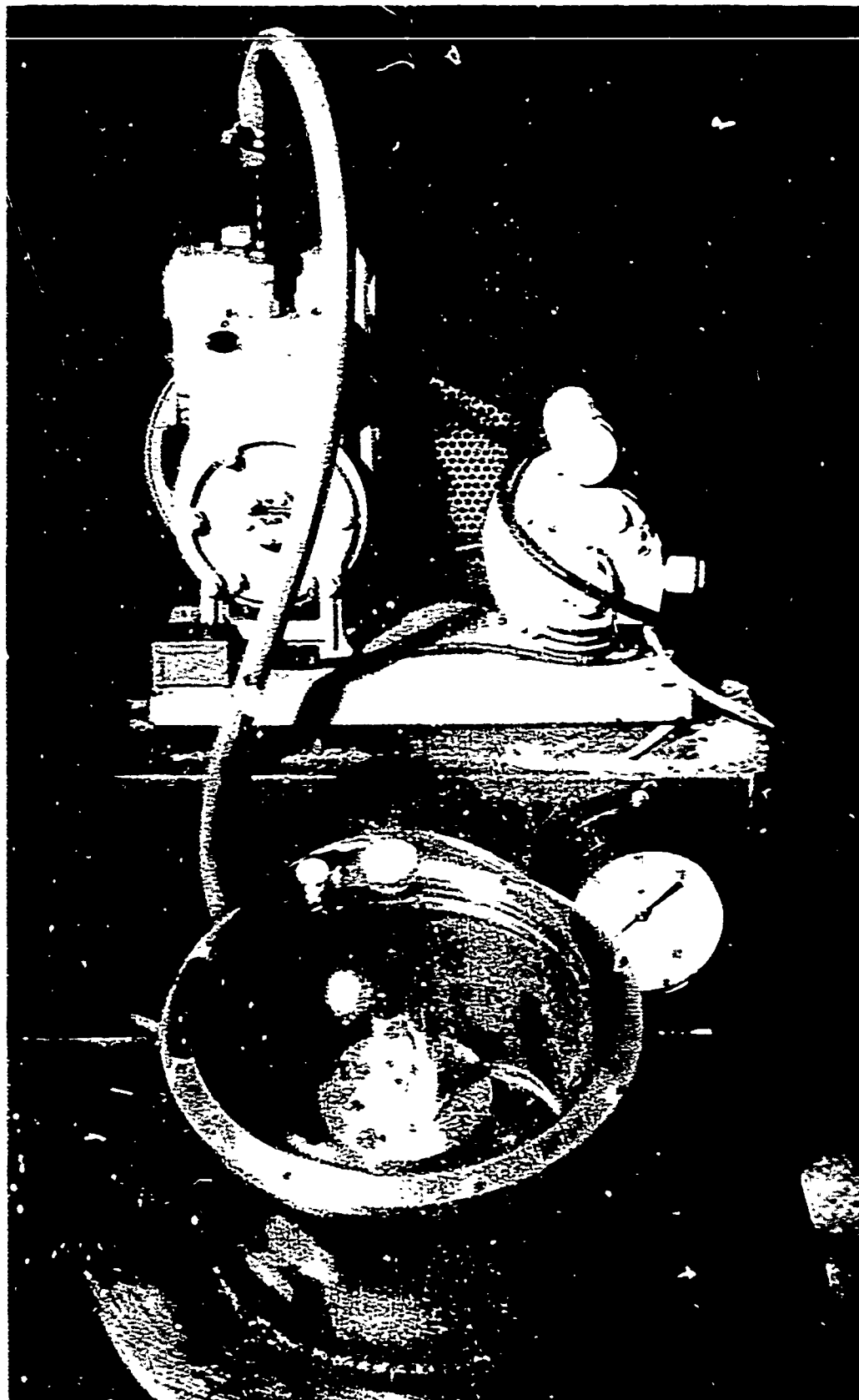


Fig. 15 PASSIVE DELAY ACTUATOR-VACUUM SHAKE TEST SET-UP

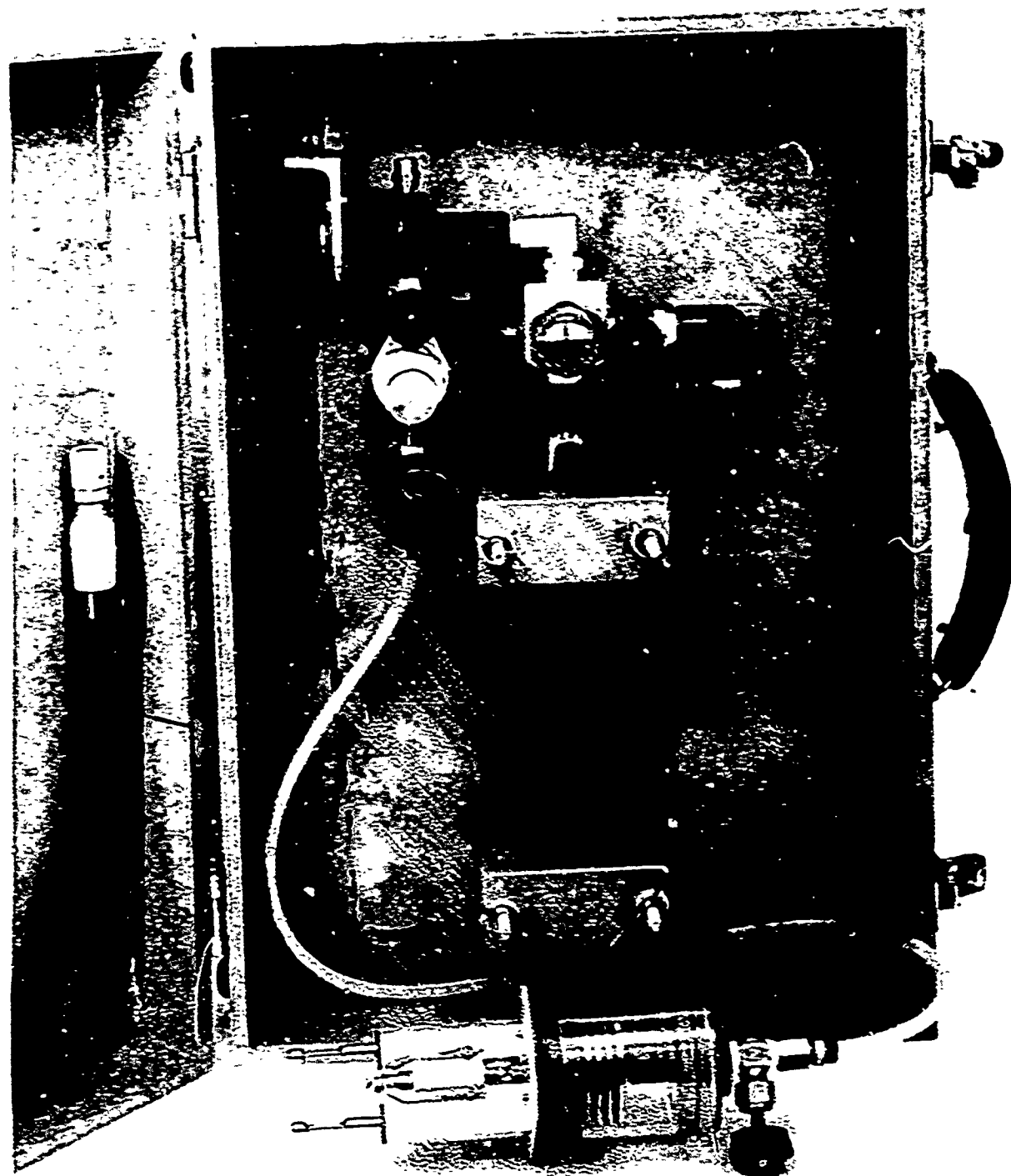


Fig. 16 PASSIVE DELAY ACTUATOR-PDA-SUITCASE TEST UNIT

to perform the necessary work cycles of the separation system, such as pulling lanyards in sequence, etc. However, it was decided early that there would be two important advantages in divorcing the force output cycles from the timing cycle of the PDA. First, by its very nature, the timing accuracy of the PDA depends to an extent on having a known and reasonably steady load on its output shaft during the timing cycle. In other words, a heavy load, such as might be caused by a tight lanyard installation, might cause the timer to run slower than if there were a light steady load. However, it has been demonstrated that the PDA is largely self compensating with respect to variable loads (as compared to steady loads). Second, and perhaps more importantly, it was decided to divorce the work output from the timing cycle so as to not only increase the magnitude of the work output, but also to permit ~~snap~~ action, which is sometimes desirable for lanyard pulling type functions. This explains the purpose of the trigger assembly that is shown as an integral part of the PDA in Fig. 17. The trigger assembly is functionally disconnected from the PDA on the ground. However during the in-flight arming cycle, the PDA output shaft is automatically connected to the trigger assembly by means of a device very similar to a hose quick-disconnect fitting. Thereafter, the trigger pawling shaft moves with the PDA shaft during the timing cycle, and releases the two (pre-cocked) triggers at precise, pre-calibrated, and adjustable time intervals. The two output shafts of the trigger assembly are, in fact, the two lanyards that release the despin and separation cables by means of the cable release assembly described in the next section.

3. The Cable Release Assembly--The spring-loaded despin weight release units are locked securely until the desired action time



Fig. 17 PDA/TRIGGER ASSEMBLY

by tying them together with a common cable. In the present separation system this cable is cut at the proper time by (redundant) explosive bolt cutters. In the simplified system, however, separate cables from each despin weight release unit are locked together at the cable release assembly, which is located directly above the trigger assembly (Figs. 18 and 19). The snap action of the despin trigger shaft causes the immediate simultaneous release of both despin cables. And similarly, the other trigger snap action causes the simultaneous release of the bolt clamp release cables.

The spring force on each cable is about 15 pounds, whereas the force on the trigger shaft required to release a pair of cables is 1 or 2 pounds. However, as stated previously, the force available from each trigger to do this work is about 23 pounds, so there is adequate safety margin. Further, the trigger that releases the bolt clamp cables is attached to the control rocket assembly for initiating the rocket firing sequence.

4. The Separation Clamp Release Bolts--The spacecraft is attached to the flight adapter by means of a split Marman clamp that is normally held in place by bolting the two halves together, thus clamping two identical flanges on the spacecraft and adapter. In the current separation system these bolts are severed at the proper time by explosive bolt cutters, which release the Marman clamp and allow a compression spring between the spacecraft and the adapter to separate the two. Since an objective of the simplified separation system was to eliminate all ordnance items, it was necessary to design a structurally equivalent bolt that would not only be capable of mechanical release, but would fit the space currently occupied by the explosive bolt cutters. After many design approaches were examined, it was decided that the

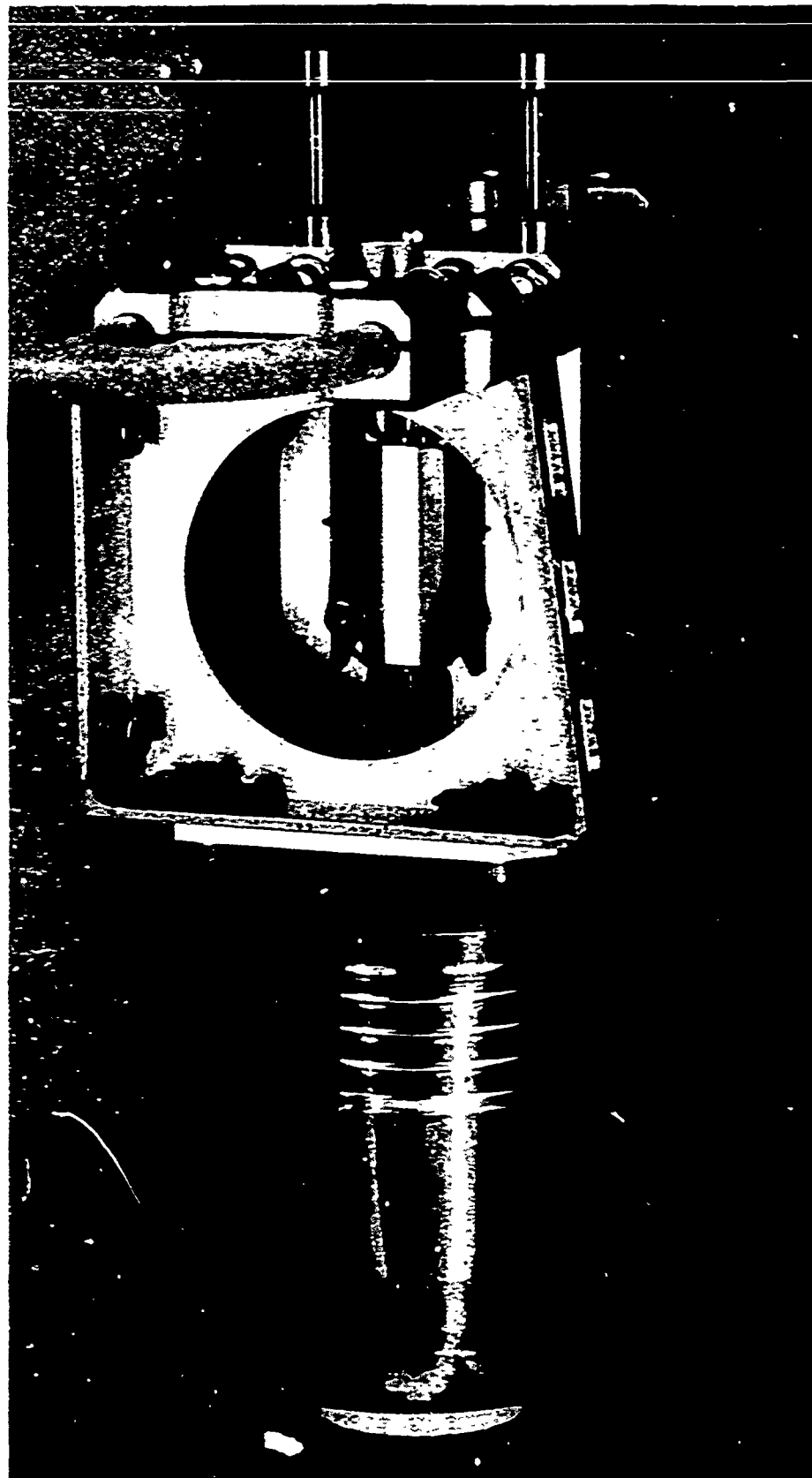


Fig. 18 PDA/TRIGGER/FLIGHT INSTALLATION

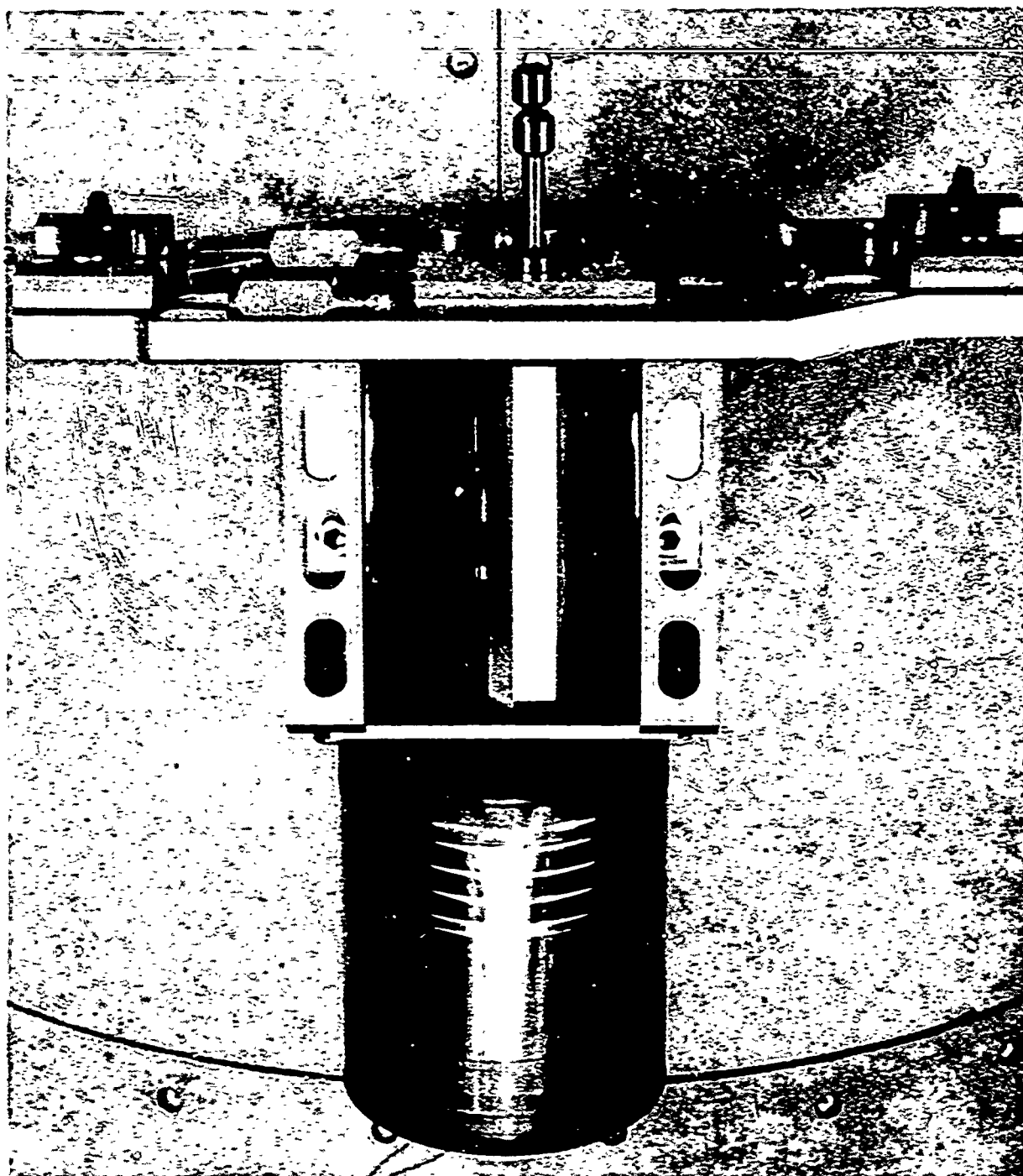


Fig. 19 PDA FLIGHT INSTALLATION SHOWING CABLE RELEASE ASSEMBLY

one shown in Fig. 20 would best suit these conditions. This arrangement is actually a miniature hinged split Marman clamp. When the two hinged halves are folded together, they grip matching flanged ends of the two bolt halves. The hinged clamp halves are then locked together by a spring-loaded cap. The cone angle of the bolt flanges and the mating surfaces of the clamp halves were carefully selected so that when the bolt halves are loaded in tension the hinged clamp halves always tend to overcome the contact friction and open. On the other hand, it was necessary to limit the surface cone angle so that the opening force on the hinged clamps would not be so great as to jam the spring loaded release cap. The prototype bolt clamp assembly is shown in the flight configuration in Fig. 21; Fig. 22 shows it released after loading the bolts to approximately 1000 pound tension

Operation of the bolt clamps is quite simple. Cables from each of the spring-loaded release cap assemblies are terminated at the cable release fixture as noted previously. Thereafter, when the "separate" trigger snaps, the cables are released, the bolt clamp caps deploy, and the clamps release. The action is virtually simultaneous and virtually instantaneous on both bolts.

By using a spring to release the clamp cap the system activation force is completely disassociated from the tension on the bolt halves. In other words, with the 15 pound spring used in the prototype bolt clamp, the activation force on the trigger shaft is essentially the same, whether the bolt tension is 700 (approximate flight spec) or 1000 pounds.

This configuration allows any desired number of tests to be conducted without degradation of the unit. The prototype bolt clamp

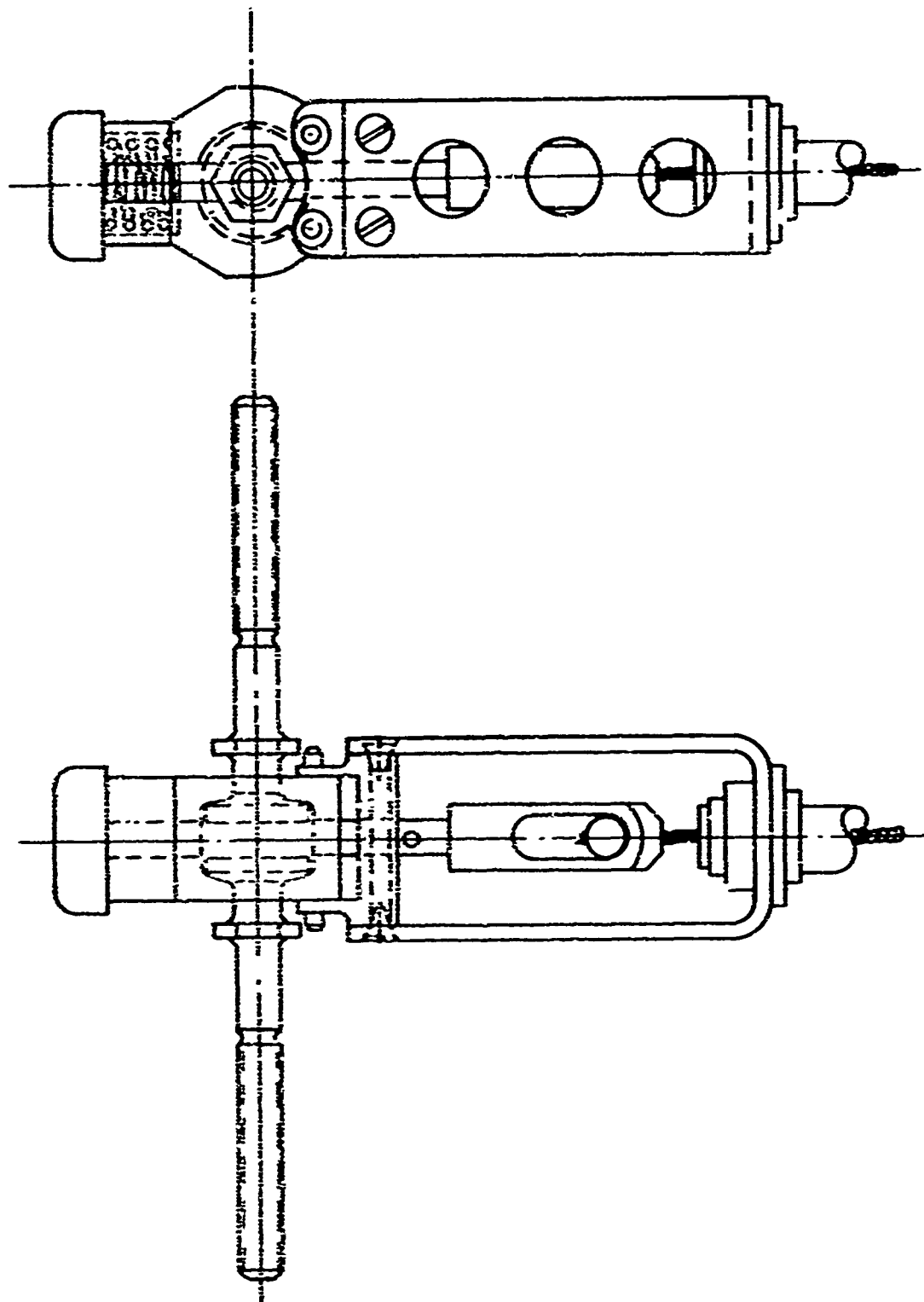


Fig. 20 BOLT CLAMP ASSEMBLY

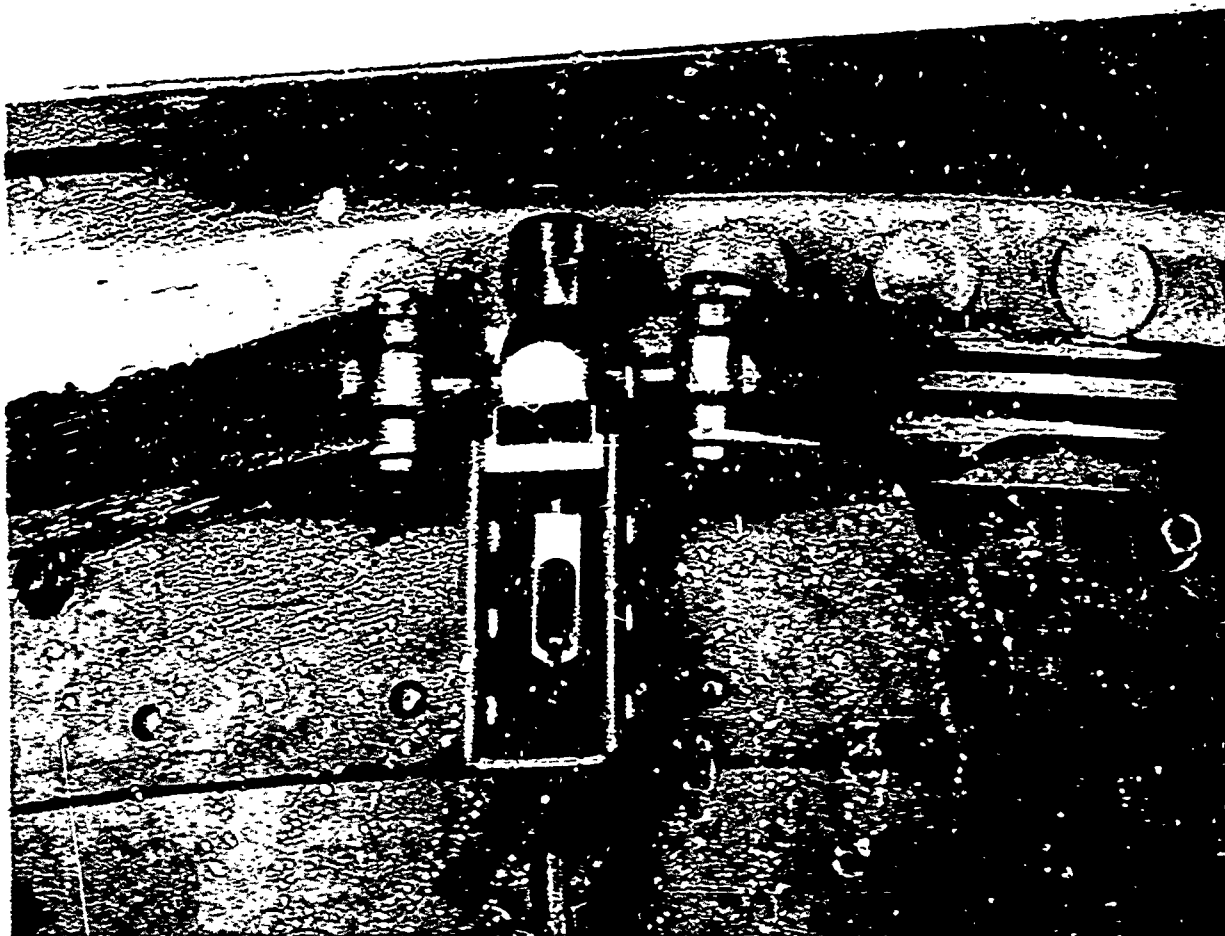


Fig. 21 BOLT CLAMP FLIGHT INSTALLATION

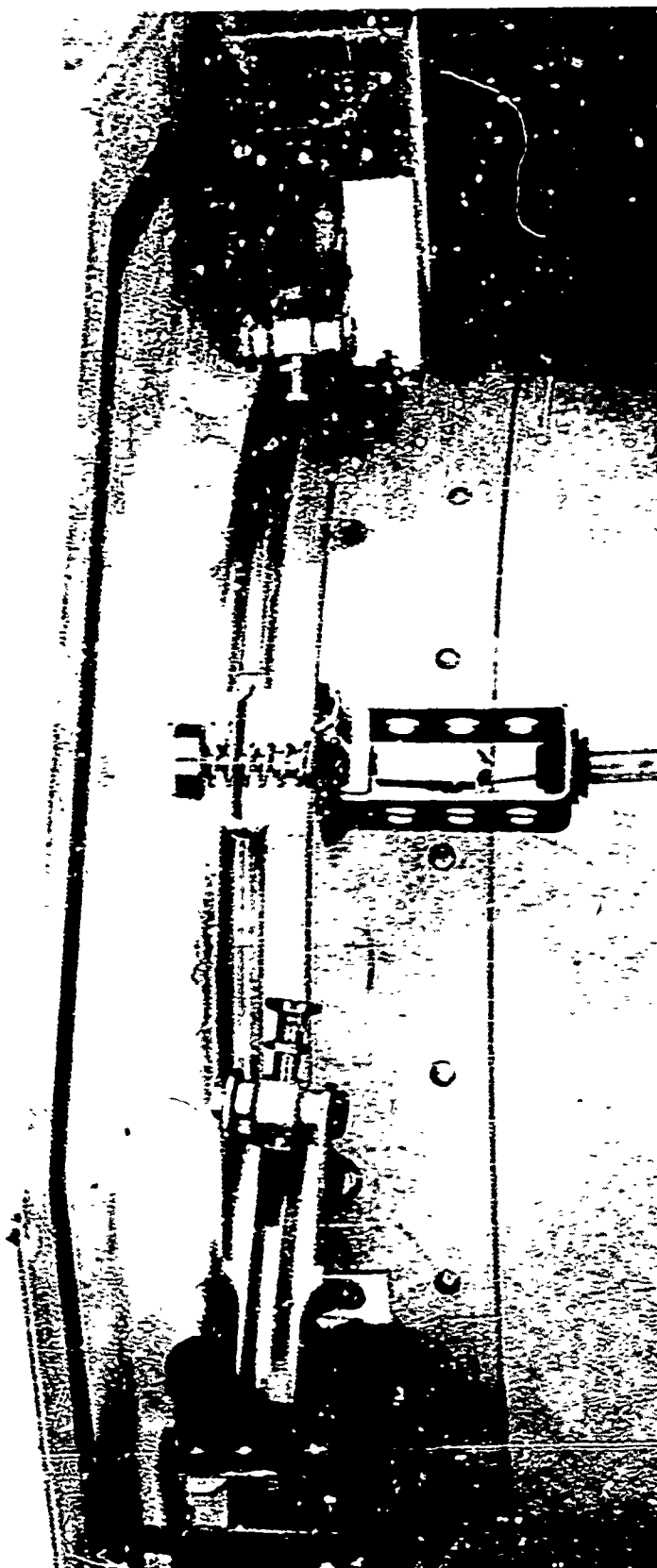


Fig. 22 BOLT CLAMP AFTER RELEASE

has been operated dozens of times in the flight configuration at 20% higher than maximum permissible flight tension loads without degradation of performance.

5. The Control Rocket--Consideration was given to many approaches to the problem of providing the necessary control rocket impulse without resorting to the use of pyrotechnics. Several of these were abandoned with the arbitrary decision that the "rocket" could not be permitted to discharge solid matter, such as slugs, springs, or pressurized cans. It was easy to demonstrate by calculation that a solid object launched transverse to the injection stage just after separation would not collide with the spacecraft in a normal separation sequence. Unfortunately, however, it was equally easy to demonstrate by calculation that the slug had a finite probability of colliding with the spacecraft in the case of an abnormal (tumbling) separation sequence.

So, the competition narrowed itself down to two acceptable approaches, both of which have been tested extensively in the Laboratory. But first it was decided to perform a complete re-evaluation of the dynamic analysis of the spacecraft/adaptor separation sequence, since it was known that some of the significant inputs to the last analysis, conducted several years previously, had changed considerably. The purpose of this analysis (see Appendix I) was to show that the required control rocket impulse was much less than that provided by the currently used 1.5 lb-sec rocket and that this is largely because the measured spin rate of several recent APL spacecrafts at injection was less than 2 RPM, whereas the previous analysis had used 30 RPM as an input. The new analysis did indicate that the required impulse to yaw the injection stage after spacecraft separation was less--by a factor of 4--so the design objective for the new control rocket was

conservatively established at 0.5 lb-sec. Actually this was only one half of the previous estimate of 1.0. The current rocket, having 1.5 lb-sec impulse, had the nearest available performance to the requirement.

The reason for the above "engineering footwork" was the (erroneous) preconclusion that an explosiveless control rocket that could produce 1.5 lb-sec of impulse would not be practical in terms of weight and volume. It has since been determined that the evaporating liquid rocket described below not only meets these requirements, but possesses the following additional advantages:

1. Hazard is virtually negligible. It contains no pyrotechnics and does not burn.
2. It does not require electrical energy to operate, which eliminates the need for the added weight and complexity of batteries, wires, connectors, etc.
3. It is immune to prefiring from RF or static electricity sources.
4. It can be fired many times in its flight configuration without hazard to itself or adjacent equipment. It can even be fired safely in the laboratory in a thermal vacuum chamber or on a satellite or launch vehicle on the launch pad.
5. Its exhaust products are nonerosive, noncorrosive, and noncontaminating.
6. It delivers about the same impulse per pound of rocket as does its solid-propellant counterpart, but is probably an order of magnitude better when the necessary solid rocket peripheral equipment is included (batteries, wires, connectors, squibs, etc.).

7. It is inexpensive. Using mass production techniques it could be manufactured for less than its solid propellant counterpart.
8. It should have a high inherent reliability. Its readiness for firing can be verified by visual inspection.
9. Its delivered impulse can be charged at any accessible time before launching (such as on the launch vehicle on the launch pad) merely by changing the amount of "fuel."
10. It can be designed for refueling in orbit, if required, but its main virtues are associated with one-shot applications.
11. Properly designed, it should be capable of indefinite shelf life without degradation.
12. Properly designed, it should be capable of satisfactory operation over a very broad temperature range, such as $\pm 100^{\circ}\text{F}$. However, future tests will be required to establish the degree of performance variation with temperature.

There are only three basic features of the evaporating liquid rocket. These are, a transparent pressure vessel, a quantity of high vapor pressure "fuel," and a releasable nozzle closure/seal.

The liquid selection is based on the best compromise of high vapor pressure, low heat of vaporization, low freezing point, ease of handling, compatibility with the pressure vessel, minimum toxicity, and, to a lesser extent, low molecular weight. Nearly any liquid will produce some impulse, when used in this manner, but the ones which best meet these requirements are the Freons, several of which have been tested with satisfactory results. Typical of these is Freon-115, which has a 70°F vapor pressure of 117 psia and a freezing temperature of -153°F .

Another is Freon-114, which has a 70°F vapor pressure of 28 psia and a freezing temperature of -137°F.

A notable feature of the rocket is that it is transparent. This is to allow visual confirmation at any accessible time that no leak has occurred since the time the rocket was loaded with fuel, whether this was 2 hours or 2 months previous. A typical rocket is shown in Figs. 23 and 24, where the container material used is Lexan. The rocket is "fired" by simply removing the nozzle closure seal, and this is accomplished in the case in question by the "separate" trigger motion. However, a nominal 2 second delay is required between the time the spacecraft separates and the time the control rocket fires, so it was necessary to provide an integral mechanical timer in the rocket closure for this purpose (not shown). Timer action is initiated by a lanyard attached to the "separate" trigger.

The Freon rocket develops its rated impulse when fired, as intended, in outer space. However, it can also be fired at any other desired time into 1 atmosphere ambient pressure for special systems tests, etc., but at a much reduced impulse. It is designed to be capable of many firings (tests) without degradation of performance.

The accurate determination of impulse delivered by the several rocket configurations tested was made possible by the unique test rig shown in Figs. 25 and 26. The invention of this test rig was necessitated by the complete futility in trying to simultaneously measure thrust (about 10 pounds) and thrust duration (about 1/10 sec) by remote control in a vacuum chamber during the initial runs. Referring to Fig. 25, the rig consists merely of a spinning arm to which the rocket is attached, a means of remotely firing the rocket after pumping the vacuum chamber down, and a frictionless means of remotely

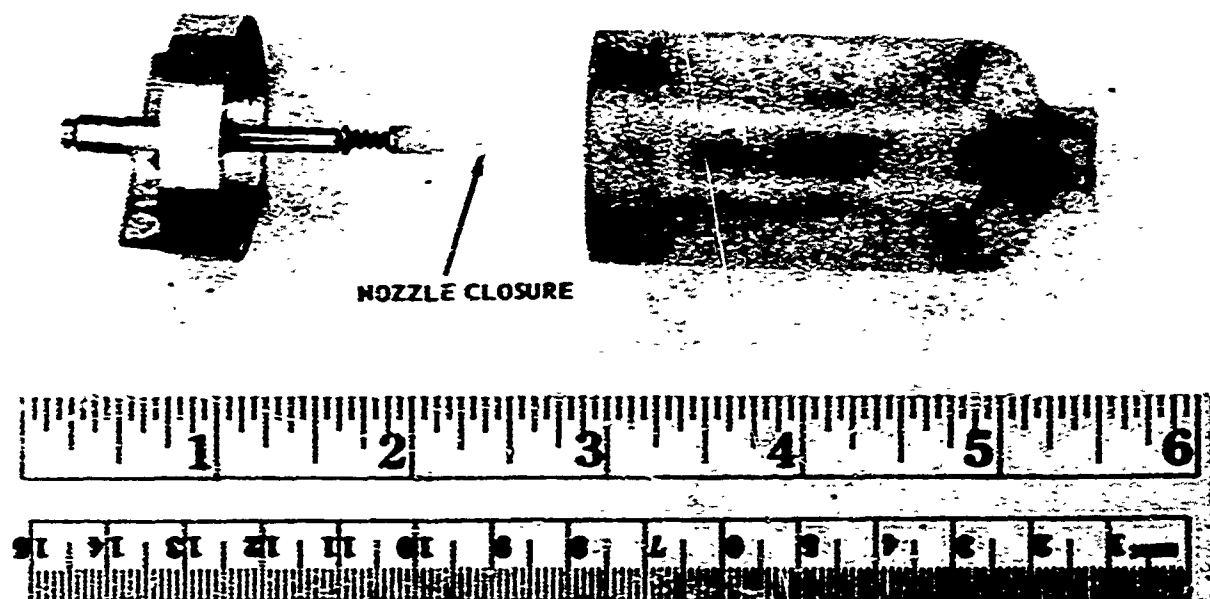


Fig. 23 PROTOTYPE TEST ROCKET HAVING $\frac{1}{2}$ LB-SEC IMPULSE

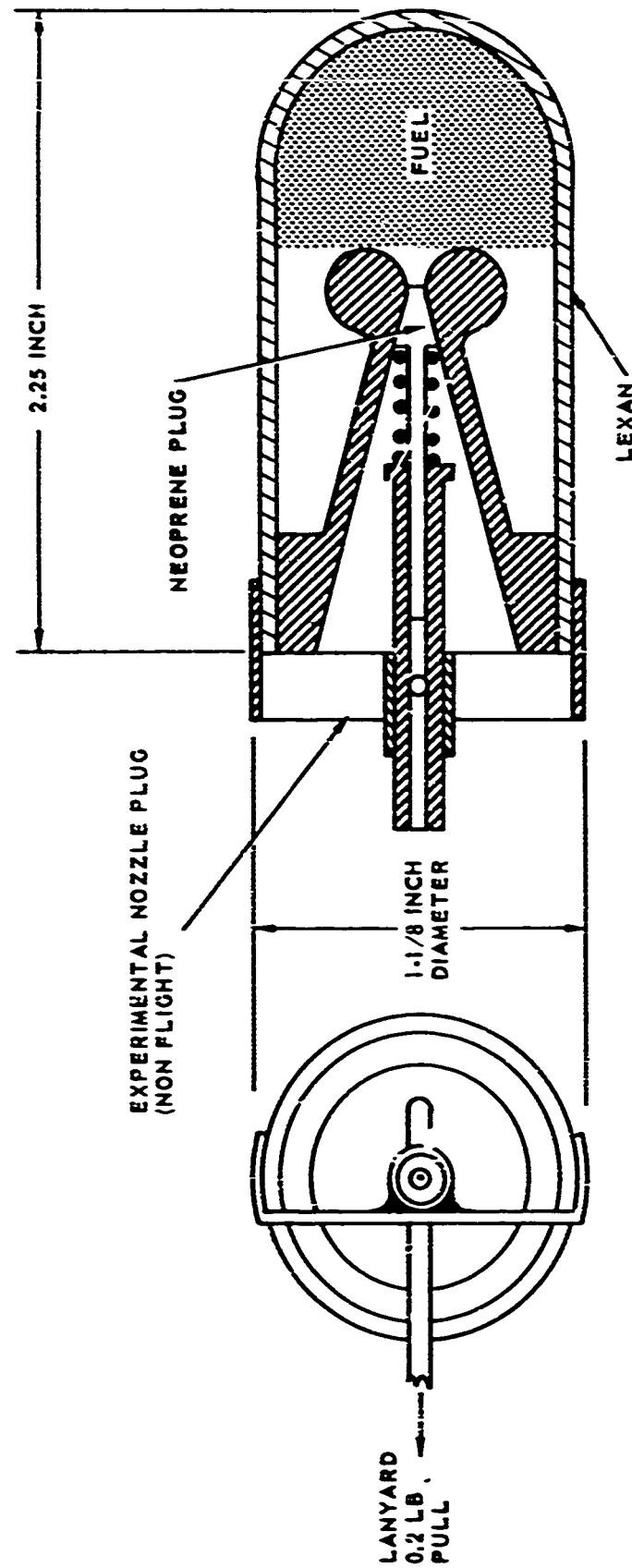


Fig. 24 EVAPORATING LIQUID ROCKET-GENERAL CONFIGURATION

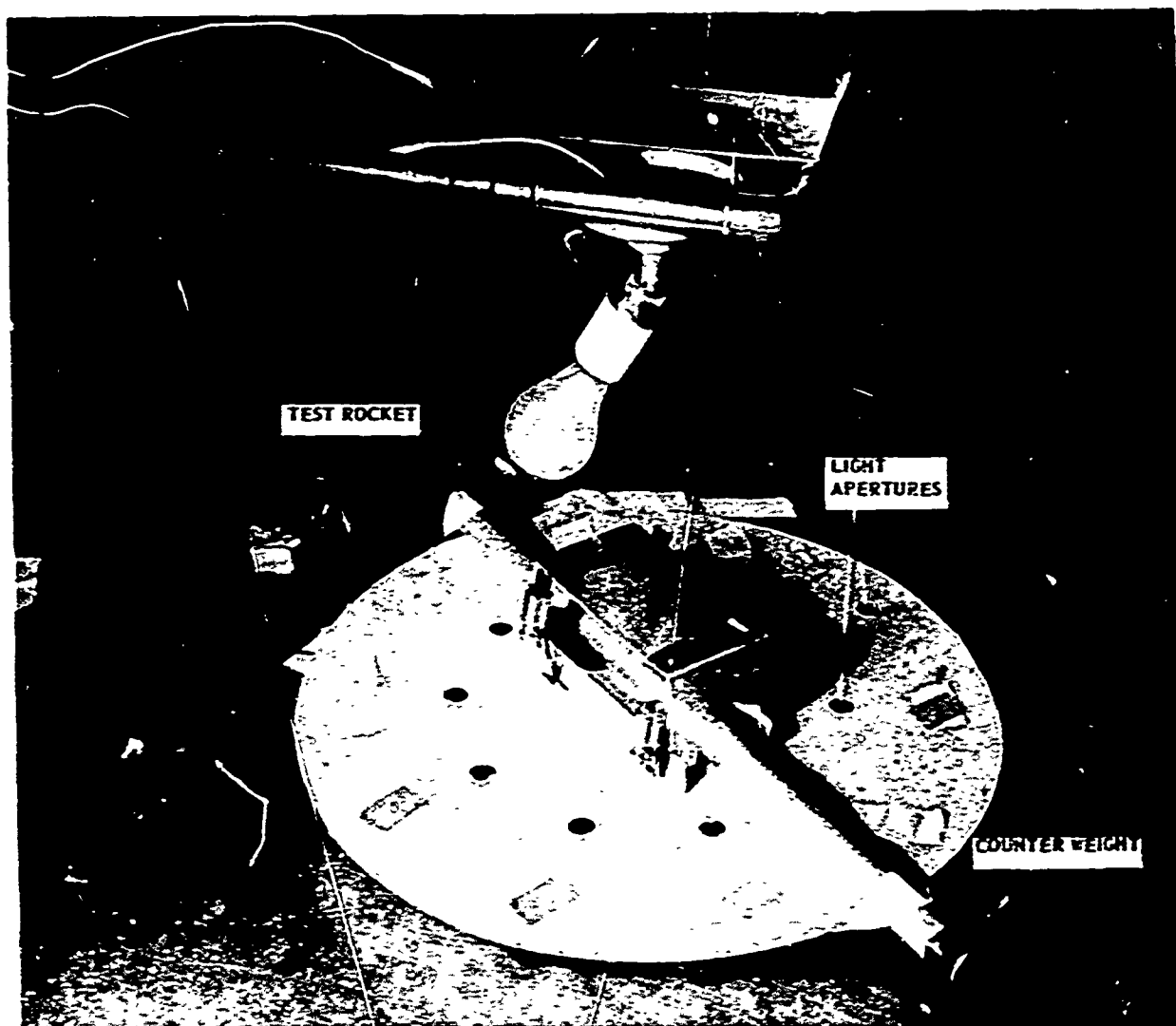


Fig. 25 CONTROL ROCKET TEST SET-UP (SPINNING EQUIPMENT)

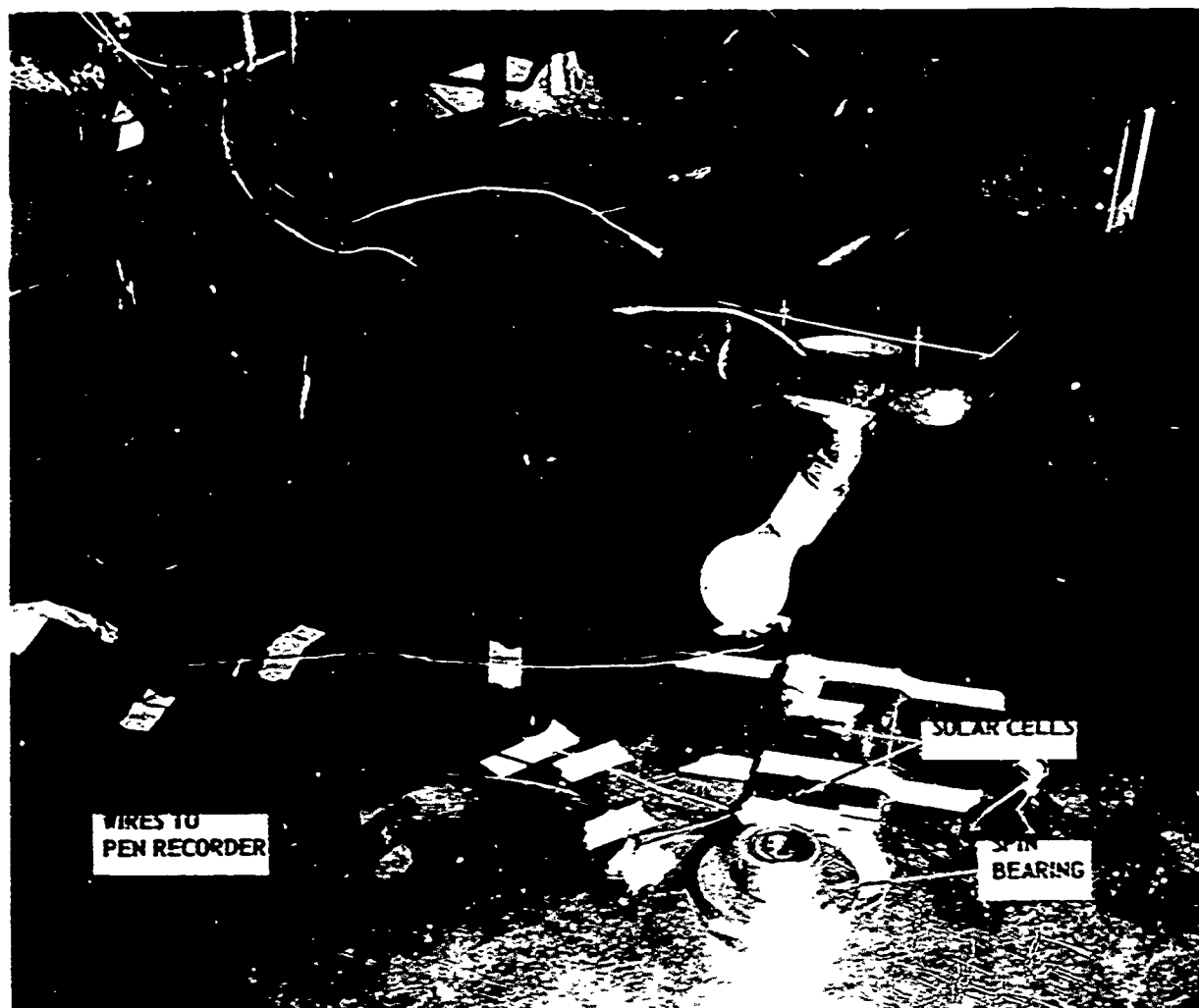


Fig. 26 CONTROL ROCKET TEST SET-UP (FIXED EQUIPMENT)

measuring the rotation rate of the arm. Firing the rocket at will was accomplished by securing the nozzle closure with string until ready for firing, and then cutting the string by remote-activated Nichrome hot-wire. Recording RPM was made equally simple by attaching to the spinning arm a disc having a hole pattern of 10 holes at one spin radius and one hole at a different spin radius. Now, by placing a light above the holes and two solar cells below the holes, and by connecting the cells and power source to a pen recorder, the RPM can be simultaneously recorded in 1/10 revolution and 1 revolution intervals.

Fueling the Freon rocket is reasonably simple, depending upon the actual fuel being used. It will be of interest to describe a typical example. In this example the fuel is Freon-114, which comes from the manufacturer in a pressure vessel that has a valve on top. At room temperature the pressure in this container is 28 psia, about 13 psig. But when the container and its contents are cooled to at least +39°F, the pressure drops to 14.7 psia, which is, of course, 0 psig. Now the valve can be opened, and the Freon-114 can be poured like any other clear, odorless liquid into the precooled rocket case. If precooling the pressure vessel and/or rocket case is not practical, the pressure vessel can be opened at 70°F, and a quantity of the Freon-114 poured while boiling into a beaker. When the boiling stops, the liquid has automatically reached +39°F, and can be poured into the rocket case. However, this procedure will cause moisture from the atmosphere to be absorbed into the Freon. If the rocket is also at 70°F, the boiling will resume until the case is automatically chilled.

In either of the above loading procedures, the nozzle closure is inserted and sealed when the proper quantity of fuel has been loaded. Thereafter, as the unit is allowed to reach ambient temperature (70°F),

the rocket internal pressure rises to 28 psia, and is ready for firing.
Should a leak develop in the rocket for any reason, it would be immediately apparent by the boiling fuel or, subsequently, low fuel level.

IV. EXTENDED APPLICATIONS OF PASSIVE SEPARATION SYSTEM COMPONENTS

There are various combinations of the foregoing components that can be used either in passive separation systems or in many other types of flight applications. The PDA can be used as a simple timer in an existing separation system, or it could be used as a timer to initiate antenna erection, boom deployment, high voltage equipment turn-ON, cover removal, etc. Similarly, the bolt clamps could be used effectively in a variety of mechanical operations where the use of explosives is undesirable. Further, there are many operations that require small one-shot remote rockets, but for many reasons cannot tolerate the heat, contamination, or power requirements of conventional types.

This document describes a series of components that have been developed for a specific application. However, all of them are scalable, although the degree of scalability has not been established for each unit nor has this been within the scope of this development program. Of particular interest in this regard, however, is one parallel study to determine the extent to which the timing cycle of a device similar to the PDA could be feasibly extended. Some isolated evaluation tests were conducted, and a number of configurations were established. These indicate that a potential increase in timing cycle of about three orders of magnitude is feasible and that it would not be unreasonable to attempt to build a reliable lightweight timer based upon these principles but which would have a timing cycle of one year or more with an accuracy of about $\pm 1\%$.

APPENDIX I

Expended X-258 and FW-4 Tipover Requirements under Low Residual Spin Conditions

The purpose of this Appendix is to show that a lower angular impulse than that presently employed is adequate to ensure safe payload separation from either the expended X-258 or FW-4 with low residual spin. It is shown that an angular impulse of 0.50 ft-lb-sec provides adequate clearance subsequent to separation either from an X-258 or FW-4 possessing 4 RPM residual spin when the tipover device is operated 2 seconds after separation.

In Ref. 1 it was shown that an angular impulse of 4.5 ft-lb-sec was satisfactory for a final stage (Scout or Delta) tipoff system, and it was proved analytically that collision between the 5A and an X-248 could not be caused by residual rocket thrust subsequent to burnout. Key parameters applicable to the earlier case were:

1. 30 RPM residual spin (The condition of zero spin was examined in Ref. 1, but the 4.5 ft-lb-sec angular impulse requirements were based on 30 RPM residual spin.)
2. 0 - 1.3 pounds residual thrust to 1600 seconds subsequent to burnout
3. Time delay for initiation of rocket tipover device 1 second (A variety of time delays were explored.)
4. X-248 mass properties
5. A requirement to displace the angular momentum vector of the rocket case 45° with tipoff

Some of the parameters applicable to the tipover problem have changed since the very complete treatment of the 5A/X-248 situation given in Ref. 1. The parameters listed below are currently applicable.

1. 0 - 4 RPM residual spin (0 - 0.42 radians per second; data from actual APL shots indicate that spin is always reduced to less than 2 RPM.)
2. 0 - 1.0 pound residual thrust
3. 2-second time delay
4. 3-ft/sec separation velocity
5. A requirement to displace the angular momentum vector of the expanded case 45° (An angular displacement of 71° is also examined.)
6. The following expended rocket parameters were obtained from Refs. 2 and 3:

	X-258	FW-4	"D" SECTION
Weight (pounds)	67	52.5	17.5
C.G. (inches aft of payload mounting shoulder)	26.6	27.71	
Residual thrust (pounds)	~ 1	~ 1	
Time of Residual Thrust	No data past 120 seconds (1 pound at that time)		
I_{roll} (slug-ft ²)	0.58	0.6	0.4
I_{pitch} (slug-ft ²)	4.2	3.8	

Calculated adapter contributions to the X-258 and FW-4 roll and pitch moments are:

X-258:

$$I_{\text{pitch}} = 2.05 \text{ slug-ft}^2$$

$$I_{\text{roll}} = 0.2 \text{ slug-ft}^2$$

FW-4:

$$I_{\text{pitch}} = 2.06 \text{ slug-ft}^2$$

$$I_{\text{roll}} = 0.2 \text{ slug-ft}^2$$

An analysis of data contained in Ref. 4 indicates that the Scout upper "D" section center of mass is approximately 8 inches aft of the fourth stage motor attachment ring. The calculated upper "D" pitch moment of inertia is 0.3 slug-ft^2 . The upper "D" pitch moment contribution to either the expended X-258 or FW-4 is then approximately 1.5 slug-ft^2 .

So that the following apply to the expended cases with payload adapter and Scout upper "D" section attached:

X-258:

$$I_{\text{pitch}} = 7.75 \text{ slug-ft}^2$$

$$I_{\text{roll}} = 1.18 \text{ slug-ft}^2$$

FW-4:

$$I_{\text{pitch}} = 7.36 \text{ slug-ft}^2$$

$$I_{\text{roll}} = 1.2 \text{ slug-ft}^2$$

Assuming separation occurs 15 minutes after injection ($\sim 50^\circ$ motion over the earth from tangential injection), the expended case is pushed downward and backward with a velocity increment of slightly less than 2 ft/sec relative to the position of the combined C.G. while the payload moves forward and upward with an increment of slightly greater than 1 ft/sec. This presumes that the low residual spin is adequate to maintain orientation between despin/solar blade erection (fourth stage burnout minus 10 mins) and separation (burnout minus 15 mins).

To displace the rocket's angular momentum vector (H) an impulsive turning moment (M), assumed instantaneous, is applied perpendicularly to the rocket spin axis. The angular momentum vector will be displaced in a direction perpendicular to that of the impulsive force and in the direction of spin. If it is planned to displace the original angular momentum vector by an amount

$$\theta = \tan^{-1} \frac{M}{H} \quad (1)$$

Initial motion will be perpendicular to both the \vec{H}_1 and $\vec{F}(t)$ vectors shown on Fig. A-1. The motor case will begin to describe the

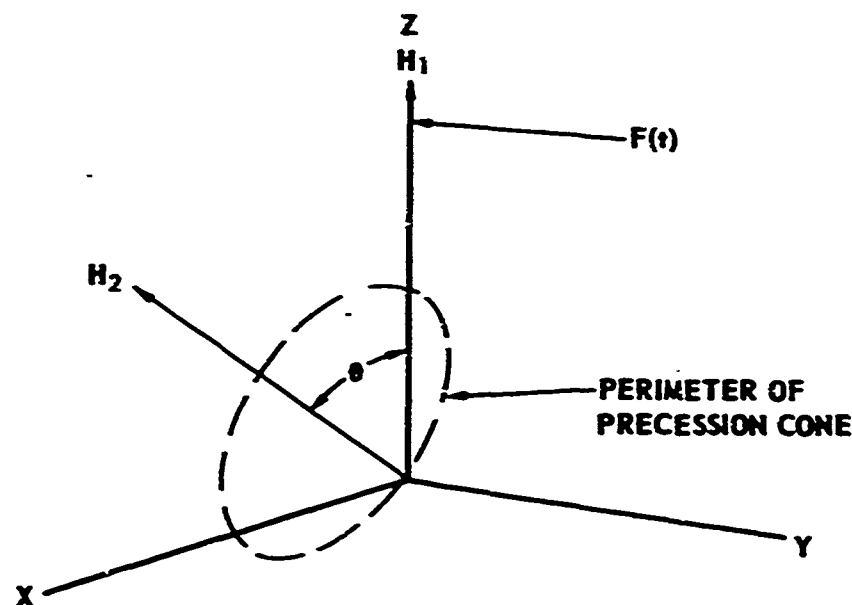


Fig. A1 INITIAL MOTION

precession cone shown with apex angle 2θ (Chapter V, Ref. 5).

Using Eq. 1 and setting $\theta = 45^\circ$ (which was an original program requirement):

$$M = H = I_{\text{roll}} \omega_z$$

$$M_{\text{X-258}} = (113) (0.42) = 0.496 \text{ ft-lb-sec}$$

$$M_{\text{FW-4}} = (1.2) (0.42) = 0.505 \text{ ft-lb-sec}$$

since $M = F \times l$ minimum total impulse requirements are

$$F_{\text{X-258}} = \frac{0.496}{3.5} = 0.142 \text{ lb-sec}$$

$$F_{\text{FW-4}} = \frac{0.505}{3.5} = 0.144 \text{ lb-sec}$$

$$F = \text{total impulse}$$

$$l = \text{longitudinal distance from the tipover device line of action to the expended rocket C. G. (approximately 3.5 feet).}$$

The spin axis of a nearly despun X-258 or FW-4 will be tilted in a direction perpendicular to the instantaneous impulsive force that caused the tilt. The resultant body motion will be precession about an axis which is the displaced angular momentum vector discussed previously and shown as vector \vec{H}_2 on Fig. A-1. The apex angle of the precession cone will be 2θ . The reason for tilting the expended rocket is to ensure that residual thrusting after burnout does not cause the case to be accelerated back into the payload. After tipover the net result of residual thrusting will be to produce an integrated motion of the expended case in the direction of the displaced angular momentum vector (\vec{H}_2).

The impulsive tipover device will also produce a lateral velocity component because this is not a pure couple. Transverse components are defined (Ref. 1):

$$V_x = \int_0^{t_1} \frac{F(t)}{M} \sin \omega_z t dt$$

$$V_y = - \int_0^{t_1} \frac{F(t)}{M} \cos \omega_z t dt$$

where t_1 = duration of pulse

Integrating and determining total transverse component by means of the expression $V_t^2 = V_x^2 + V_y^2$ results in:

$$V_t^2 = 2 \left(\frac{F}{M \omega_z} \right)^2 [1 - \cos \omega_z t]$$

At least 90 percent thrust is developed in the first second with the practical passive tipover devices tested thus far so that a minimum lateral velocity component would be:

$$\begin{aligned} (V_t)^2 &= 2 \left(\frac{0.14}{2.77 \times 0.42} \right)^2 \left(1 - \cos [0.42 \times 57^\circ (1)] \right) = (0.021 \text{ ft/sec})^2 \\ V_t^2 &= 0.145 \text{ ft/sec} \end{aligned} \quad (2)$$

Residual thrusting will accelerate the expended rocket toward the payload subsequent to separation. If the rocket and payload centers of mass are initially separated by an amount S_0 , an expression which specifies the distance between centers up to operation of the tipover device is:

$$S = S_0 + (\Delta v)t - \left(\frac{T_r}{M} \right) \frac{t^2}{2} \quad (3)$$

By using this equation with the following initial conditions it is possible to show that collision could occur at a relative velocity of 3 ft/sec 16.5 seconds after separation without tipover.

ΔV (separation velocity)	=	3 ft/sec
T_r (residual thrust)	=	1 pound
M :		
Adapler	=	0.22 slug
Case	=	2.0 slug
Upper D	=	0.55 slug
Total Mass (M)	:	2.77 slug

Subsequent to separation there is a short time delay (~ 2 seconds) before the tipover device is operated. This allows the directional antenna to unfold and clear. When the tipover device operates, the expended rocket spin axis is cocked causing residual thrusting to impart a transverse acceleration to the expended rocket.

Two cartesian coordinate systems are shown on Fig. A-2. The x axis is parallel to the original separation velocity vector. The $\bar{x} \bar{y} \bar{z}$ system is rotated so that the \bar{z} axis aligns with the displaced angular momentum vector. The tipover impulse vector was therefore parallel

to the $y(\bar{y})$ axis. The expended rocket now processes about the \bar{z} axis with the simple harmonic motion shown projected to the $\bar{x} - \bar{y}$ plane on Fig. A-2 (Ref. 4):

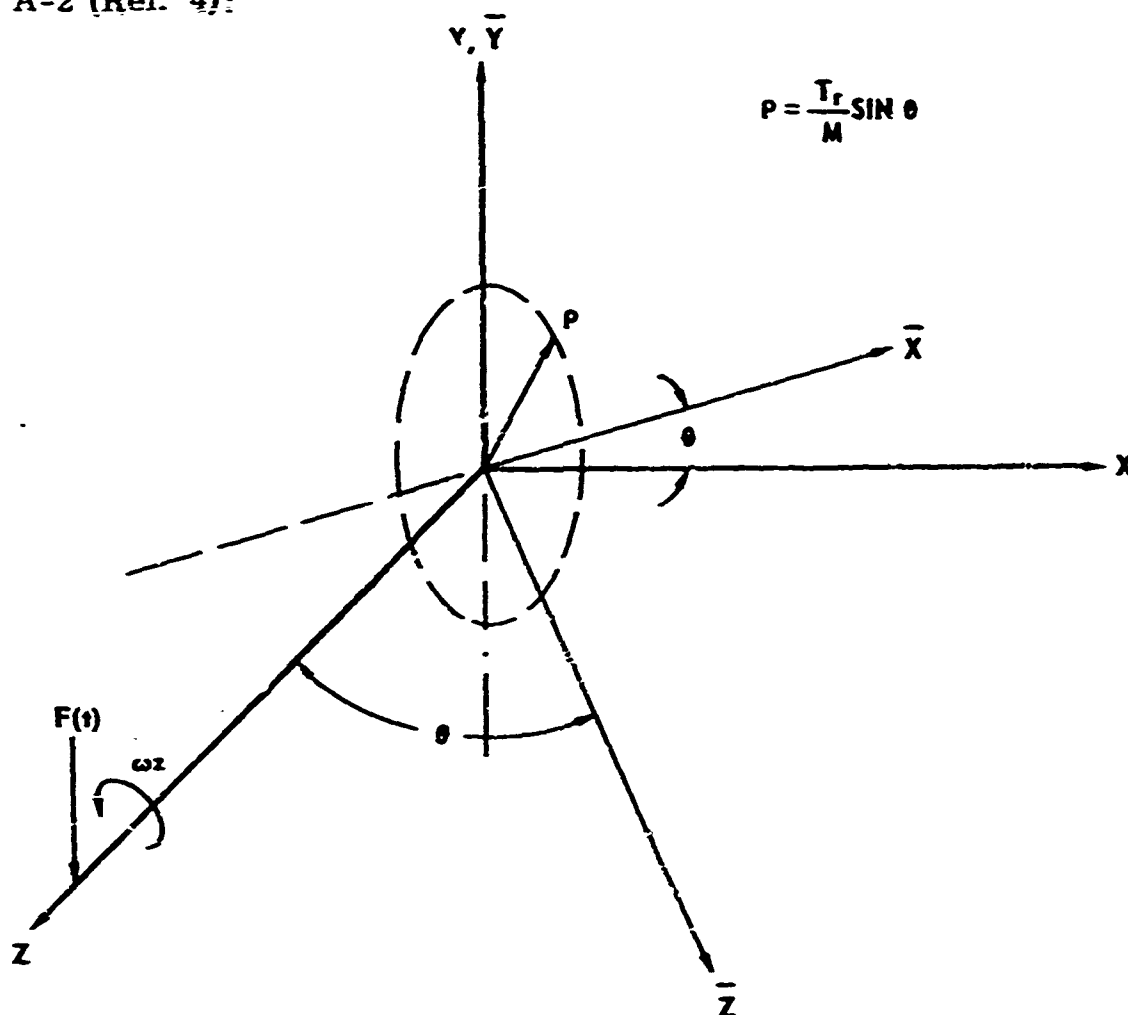


Fig. A2 ROCKET PRECESSION

P is the acceleration component normal to the \bar{z} axis.

Acceleration components in the $\bar{x} \bar{y} \bar{z}$ system subsequent to tip-over are:

$$\ddot{\bar{x}} = -P \cos \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right),$$

$$\ddot{y} = \rho \sin \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right),$$

$$\ddot{z} = \frac{T_r}{M} \cos \theta$$

Components in the $\bar{x} \bar{y} \bar{z}$ system are transformed to the $x y z$ system by the matrix

$$\begin{vmatrix} \cos \theta & 0 & \sin \theta \\ 0 & 1 & 0 \\ -\sin \theta & 0 & \cos \theta \end{vmatrix}$$

So that acceleration components in the $x y z$ system caused by residual thrusting after tipover may be written:

$$\ddot{x} = -\rho \cos \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right) \cos \theta + \frac{T_r}{M} \cos \theta \sin \theta$$

$$\ddot{y} = \rho \sin \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right)$$

$$\ddot{z} = \rho \cos \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right) \sin \theta + \frac{T_r}{M} \cos^2 \theta$$

Expressing radial acceleration by the relationship $\ddot{R}^2 = \ddot{X}^2 + \ddot{Y}^2$:

$$\ddot{R}^2 = \rho^2 \left[\cos^2 \theta \left(\cos \left[\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right] - 1 \right)^2 + \sin^2 \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right) \right]$$

After using the half angle relationships, the binomial expansion, and neglecting high order terms, radial acceleration may be approximated by:

$$\ddot{R} \approx \frac{T_r}{M} \sqrt{3} \sin \theta \sin \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right) \quad (4)$$

The z axis acceleration components is written:

$$\ddot{z} = \frac{T_r}{M} \left[\cos^2 \theta + \cos \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_{zt} \right) \sin^2 \theta \right] \quad (5)$$

This longitudinal (along z axis) and transverse (radial) displacement of the expended rocket caused by thrusting after tipover is obtained by integrating Eqs. 4 and 5 twice with respect to time. Velocity components caused by thrusting in the first procession cycle after tipover are:

$$\begin{aligned} \dot{R} &= \frac{T_r}{M} \sqrt{3} \sin \theta \left(\frac{2I_{\text{pitch}}}{I_{\text{roll}} \omega_z} \right) \left(1 - \cos \left[\frac{I_{\text{roll}}}{2I_{\text{pitch}}} \omega_z (t - t_e) \right] \right) \\ \dot{z} &= \frac{T_r}{M} \left([t - t_e] \cos^2 \theta + \frac{I_{\text{pitch}}}{I_{\text{roll}} \omega_z} \sin^2 \theta \sin \left(\frac{I_{\text{roll}}}{I_{\text{pitch}}} \omega_z [t - t_e] \right) \right) \end{aligned}$$

and, displacements are:

$$R = \frac{T_r}{M} \sqrt{3} \sin \theta \left(\frac{2I_{\text{pitch}}}{I_{\text{roll}} \omega_z} \right) \left([t - t_e] - \frac{2I_{\text{pitch}}}{I_{\text{roll}} \omega_z} \sin \left[\frac{I_{\text{roll}} \omega_z}{2I_{\text{pitch}}} [t - t_e] \right] \right) \quad (6)$$

$$z = \frac{T_r}{M} \left(\frac{[t - t_e]^2 \cos^2 \theta}{2} + \left[\frac{I_{pitch}}{I_{roll} \omega_z} \right]^2 \sin^2 \theta \left[1 - \cos \left(\frac{I_{roll} \omega_z}{I_{pitch}} [t - t_e] \right) \right] \right) \quad (7)$$

Equations 2, 3, 6, and 7 are now combined to produce the following two expressions for longitudinal and transverse displacement as a function of time from separation to completion of the first precession cycle after tipover.

$$R = \left(V_t \right) [t - t_e] + \frac{T_r}{M} \sqrt{3} \sin \theta \left(\frac{2I_p}{I_R \omega_z} \right) \left[[t - t_e] - \frac{2I_p}{I_R \omega_z} \sin \left(\frac{I_R \omega_z}{2I_p} [t - t_e] \right) \right] \quad (8)$$

$$z = S_0 + (\Delta V) t - \frac{1}{2} \left(\frac{T_r}{M} \right) t_e^2 - \left(\frac{T_r}{M} \right) \left[t_e [t - t_e] - \left(\frac{T_r}{M} \right) \frac{[t - t_e]^2 \cos^2 \theta}{2} + \left(\frac{I_p}{I_R \omega_z} \right)^2 \sin^2 \theta \left(1 - \cos \left(\frac{I_R \omega_z}{I_p} [t - t_e] \right) \right) \right] \quad (9)$$

where

t_e = event time (tipover)

T_r = residual thrust (1 pound)

M = rocket mass (2.77 slugs)

ΔV = initial separation velocity (3 ft/sec)

S = distance between rocket and payload C. G. 's prior to separation

S_o = 5 feet = separation distance between rocket and payload mass centers before separation

V_t = transverse velocity increment caused by tipover rocket thrust

I_p = expanded rocket pitch moment of inertia (7.5 slug-ft²)

I_R = expanded rocket roll moment of inertia (1.2 slug-ft²)

θ = angular displacement to rocket spin angular momentum vector

ω_z = residual spin rate (0.42 rad/sec)

Equations 8 and 9 define the separation distance between the rocket and payload center of mass subsequent to separation.

Radial separation as a function of time is plotted on Fig. A-3 for the following cases:

1. 0.5 ft-lb-sec angular impulse using: $t_e = 2$ seconds,
 $I_{pitch} = 7.5$ slug-ft², $I_{roll} = 1.2$ slug-ft².
2. 1.75 ft-lb-sec angular impulse with other parameters of 1 above.

The Z intercept time of Fig. A-3 denotes the point where $z = 0$. It is desirable to tilt the rocket as soon as possible after separation so as to increase lateral separation quickly. A 2-second time delay has been used here to allow the directional antenna to clear. Eight feet is the minimum acceptable separation as time increases (Ref. 1). The precession period is 93.5 seconds. Clearance at the completion of the first half cycle in precession is greater than 300 feet for both cases 1 and 2 above.

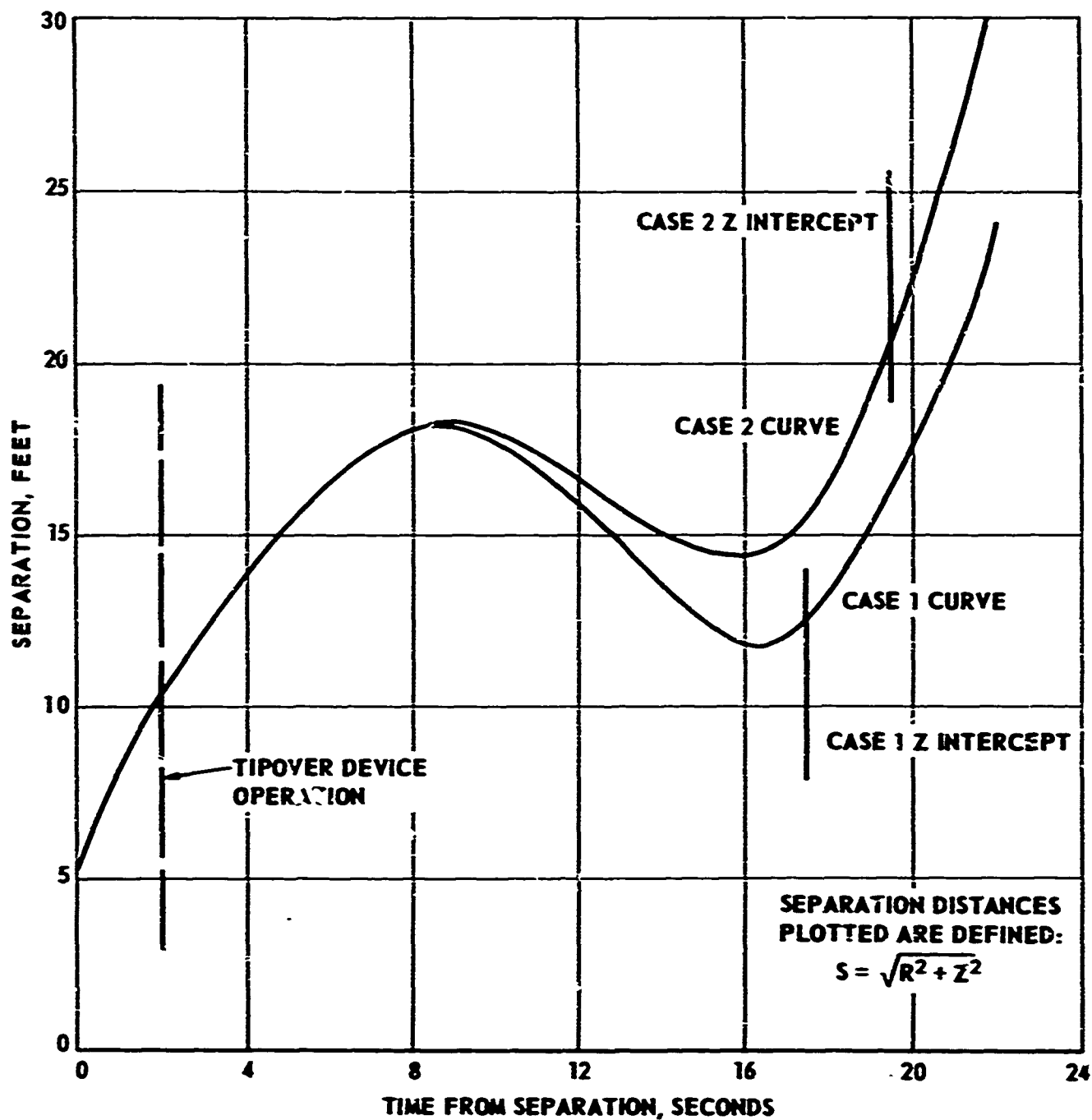


Fig. A3 SEPARATION BETWEEN PAYLOAD AND ROCKET CENTERS OF MASS

SUMMARY

This appendix shows that a total impulse of either 0.14 or 0.5 lb-sec directed radially into the spin axis of either the X-258 or FW-4 for a moment arm of 3.5 feet from the expended rocket C.G. is adequate to insure that no subsequent collision of the case with the separated payload can occur due to residual rocket thrusting in the event spin orientation is retained for the five minute period subsequent to despin/blade erect and that the separation impulse is then symmetric with respect to both the payload and rocket mass distribution. Such preservation of orientation seems unlikely since the final stage is always nearly completely despun.

The purpose here has been to show that a substantial reduction in angular impulse from that presently employed appears feasible. As is shown on Fig. A-3, a minimum impulse (0.14 pound-second) produces a 12 foot close approach (CM) clearance with a residual spin of 4 rpm. A higher tipover impulse improves matters as would a lesser residual spin rate.

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13. ABSTRACT <p>APL spacecraft require equipment: (1) to initiate despin and solar blade unfolding; (2) to separate the spacecraft from the launch vehicle injection stage; (3) to yaw the injection rocket after separation to prevent collision with the spacecraft. This report describes a unique simplified separation system which accomplishes the same objectives, but has the following advantages over the method presently used: (1) functional components are 60% lighter; (2) it has no batteries or wiring; (3) it requires no ordnance of any kind; (4) it is immune to RF, static, or other electrical background disturbances; (5) its environmental temperature limitations are far in excess of similar limitations on the spacecraft and the launch vehicle; (6) it has indefinite shelf life without servicing; (7) its operation is completely independent of the launch vehicle configuration; and (8) it can be tested and operated at any place any desirable number of times without hazard to itself or adjacent personnel or equipment.</p>			

14.

KEY WORDS

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